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HUGHES TOOL COMPANY · AIRCRAFT DIVISION
Culver City, California

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CONTRACT NO. AF 33(600)-30271

# HOT CYCLE ROTOR SYSTEM
# ENGINE-ROTOR CONTROL STUDY

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HUGHES TOOL COMPANY -- AIRCRAFT DIVISION
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# SECTION 1

## SUMMARY

In accordance with Item 12 of Air Force Contract AF 33(600)-30271, (D/A Project Number 9-38-01-000, Subtask 616) a study has been made of the engine-rotor control system for a hot cycle rotor powered by two General Electric T64 gas generators. The study included a comparison of the techniques currently used for governing free turbine turboshaft engines both in single and dual installations. All of the free turbine turboshaft engines investigated, including the T64, have droop-stabilized governors to provide essentially constant rotor rpm. Each governor has within it two flyball-type governors, one to measure and control free turbine or rotor speed, and the other to measure and control gas generator speed or engine power. The value of governed rotor speed is selected by the pilot by movement of a cockpit rotor speed selector lever which changes the reference load on the free turbine speed governor. Engine output power is selected indirectly by the aircraft controls by means of increasing or decreasing the load on the engine (i.e., helicopter rotor collective pitch). The free turbine or rotor speed governor controls the gas generator speed governor to match engine power to load power and thus maintain rotor speed essentially constant. The flyball type governors employed are simple and very reliable, but they permit a small change of rotor rpm with load. This speed change is felt as a reduction of 6-10% in governed rotor rpm as collective pitch is increased from idle load to full power. This speed decrease is known as "droop" and is generally removed by resetting the reference rotor speed in proportion to the change of collective pitch to maintain essentially constant rotor rpm.

It was found that the present control system for the YT64 turboshaft engine (the engine model available for test) can successfully govern the hot cycle rotor with no modification in the basic governing circuitry and mechanism. The dynamic behavior of this system for load disturbances, frequency response, behavior after one engine failure, and for power recovery from practice autorotation will be similar to or better than that of other current free turbine turboshaft engine installations.

A design study was also made of the one engine out condition to establish the requirements for diverter valves and blade duct valves to permit operation with one engine out. It was found that one engine operation can be readily obtained. Three gas generator-diverter valve configurations, six diverter valve designs, and two blade duct valve designs were studied.

Many combinations of these components can be used successfully with little or no effect on the dynamic performance of the control system. A detailed discussion of the operation of the over-all system is given. Some operations are necessarily automatic; others can safely be performed by the pilot.

A brief test program is proposed of pre-whirl tests of the T64 gas generators on the whirl tower. These tests will demonstrate feasibility of the control concepts proposed here.

## SECTION 2

## INTRODUCTION

This report summarizes an engine-rotor control study done under Item 12 of Air Force Contract AF 33(600)-30271, Hot Cycle Rotor System. The objectives of Item 12 are to clearly define the engine-rotor control requirements and then to make a preliminary design of a control system to satisfy those requirements. The contract required that the study be divided into the following two phases of work:

a. A comparison of the techniques for governing free-turbine turboshaft engines both in single and dual installations to establish what, if any, modifications are required in the basic governing circuiting and mechanism for application to the Hot Cycle.

b. A study of the one engine out conditions for the Hot Cycle Rotor powered by two T64 gas generators. This study establishes the requirements for selector valves and check valves to permit operation with one engine out. Preliminary design is to be made of the required valves and seals. This study shall also establish whether valve actuation shall be automatic or pilot operated.

Preliminary investigation revealed that reliable evaluation of the stability and response of an engine-rotor control system can only be made if a specific study is made of the characteristics of a particular rotor and particular engines. Valuable information is available on the behavior of the engine-rotor control systems of several free-turbine turboshaft engines in single and dual helicopter installations. However, this material is general in nature and provides no assurance that some other engine-rotor combination would be a stable system.

Of particular importance is that all free-turbine installations that have flown to date exhausted the gas generator gas through a high speed power turbine immediately behind the gas generator. The lift fan powerplant of General Electric does displace the power turbine a few feet from the gas generator. But no installation to date exhausted the gas generator gas through a large diameter rotor with nozzles thirty or more feet away from the gas generators. In addition, examination of the dynamic characteristics of the gas generator and fuel control of engines such as the T63 and T64 reveal substantial differences in time constants, gains, and idle speeds.

Because of the differences between earlier installations and differences in the dynamics of various gas generators, it was felt necessary to make a complete dynamic analysis of the combination of two T64 gas generators driving the 55-foot diameter Hot Cycle rotor developed under this contract. By working with the fundamental physics of this problem, rather than by generalizing from other installations only slightly similar, it is felt that a more reliable study was produced. At the same time, a great deal was learned of the general characteristics of the T64 gas generator. This information contributed to the design study of valves, seals, and valve actuation and coordination.

Within this framework of relating the whole study specifically to the hot cycle rotor and T64 gas generator combination, the control study was divided into two phases:

Phase I Application of Current Governing Techniques to Hot Cycle - T64 System

Phase II One Engine Out Design Study

This report summarizes that study. It also includes a proposed pre-whirl test program to demonstrate the feasibility of the hardware and operating concepts derived in this study.

# SECTION 3

## PHASE I - APPLICATION OF CURRENT GOVERNING TECHNIQUES TO HOT CYCLE - T64 SYSTEM

### 3.1 INVESTIGATION OF TECHNIQUES FOR GOVERNING FREE TURBINE Turboshaft Engines

In accordance with the Scope of Work (Reference 1) defining this study, a review was made of the techniques for governing free-turbine turboshaft engines in single and dual installations. The following aircraft were investigated:

| Helicopter Designation | Number and Type of Engine | Reference |
|---|---|---|
| Sikorsky S-58/T | Two General Electric T58-GE-6 | 2 |
| Bell XH-40(HU-1) (YHU-1B) | One Lycoming T53-L-3(5) | 3, 4, 5 |
| Sikorsky S-62A | One General Electric CT58-100-1 | 6 |
| Vertol YHC-1A (Model 107) | Two General Electric T58-6E-6 | 7 |
| Vertol YHC-1B (Chinook) | Two Lycoming T55-L-3 | 8 |

#### 3.1.1 Droop-Stabilized Control

References 2 through 8 discuss five different helicopters with three different engines; the General Electric T58 and Lycoming T53 and T55. In each case, according to the references, a droop type speed control is used which changes engine fuel flow in inverse proportion to speed error.

The droop type control is a simple way of obtaining stable, so-called "constant speed" control; however, the governed speed reduces slightly as load is increased, and vice-versa. This droop type of control is basically the type used on the T64; therefore comments concerning droop in these other installations may apply to the hot cycle rotor - T64 gas generator combinations.

3.1.2 Static Droop; Static Droop Compensation

The droop type of speed control used to date by manufacturers of free-turbine engines has permitted static speed droops of up to 10% rpm for a 100% load change, as indicated in Reference 8, for example. Pilot reaction to this type of control with so much rotor speed change has been uniformly negative. Consequently, all recent free turbine installations have had an airframe-supplied static droop elimination cam of some sort to attempt to remove the objectionable static droop which the engine manufacturer uses to stabilize the system. Ideally, a set of cams can be made which would provide constant rotor rpm for all speeds, power settings, and altitudes. This arrangement is too complicated and expensive; a single cam is generally used. However, as pointed out in Reference 4, a single droop elimination cam cannot correct power turbine speed (or rotor speed) exactly over a wide range of density altitudes. Care must be used to design a cam which gives the least average error over the aircraft flight envelope. According to References 5 and 7, it is possible to reduce static droop to 2 - 3% with a single cam. However, this much residual droop is considered unsatisfactory in Reference 5, and Reference 7 recommends that static droop of 1% be obtained. Therefore, a check will be made to see if the rotor speed of hot cycle-T64 combination can be held this closely.

3.1.3 Dynamic Droop

Dynamic droop, or the temporary deviation of rotor rpm from the governed value during power transients is shown in Reference 2 to be about 5%, and in Reference 7 to be about 6%. Reference 7 recommends that dynamic droop be held to 2%. As will be pointed out later, the dynamic droop is chiefly a function of how much fuel is introduced into the engine for a given rotor speed error. This relationship of fuel to speed error is called the main "gain" of the system. It roughly corresponds to the spring of a one degree of freedom system - the stiffer the spring, the less the dynamic deflection. However, the engine-rotor systems studied here are all multiple degree of freedom systems which tend to go unstable if the main gain is too high. Therefore, for the hot cycle-T64 combination, it will be necessary to examine the transient rotor speed behavior as the main gain is changed. The objective will be to reduce dynamic droop to 2% without making an unstable system.

3.1.4 Governor or "Beep" Actuation Rate

All the helicopters of References 2, 3, 4, 5, 7, and 8 are equipped with rotor speed selector controls of the electric switch type. These switches are usually on the collective stick and are used as "beep" switches to select

a particular rotor rpm.. They are also used to correct the rotor rpm because of the deficiencies of the droop compensation cams discussed in Paragraph 3.1.2. The fuel governor actuation rate in response to these "beep" switches is criticized in References 5 and 7 as being too slow. An actuation rate of 4% change of rotor rpm in 3 seconds is recommended in Reference 7. This objective should be kept in mind if a "beep" system is used on the hot cycle. Reference 6 points out that the Sikorsky S-62A controls rotor speed manually by twisting of the conventional throttle grip. This manual system appears to be more desirable than the slow-acting electrical system, but it presents a problem of a complicated throttle for two-engine installations. Sikorsky solves this problem on the two engine HSS-2 helicopter by moving the rotor speed controls to separate levers on the overhead panel. This arrangement retains the advantage of rapid direct mechanical control of rotor speed, but it requires the pilot to take his hand from the collective stick to reset rotor rpm. It is not necessary to do this with the electrical "beep" switches on the collective stick for one or two engine helicopters. But apparently, as pointed out here, the "beep" actuation rotor must be speeded up. If it cannot, the separate manual system may be the better choice. (Probably located on a pedestal, rather than overhead.)

### 3.1.5 Drive System Instability

All of the free turbine turboshaft installations discussed in Reference 2 through 8 have an elastic drive system between the power turbine and the rotor. These drive systems all have several shafts and inertias, including speed change gear boxes which effectively multiply stiffnesses and inertias. The XH-40 (HU-1), discussed in Reference 3, 4, and 5, has teetering rotor with no drag hinges and is rigid in the chordwise direction. All the other references discuss helicopters with rotors with drag hinges and blade dampers. In all cases, these drive systems have natural torsional frequencies for one or more modes of torsional vibration. It is pointed out in Reference 3 that all fuel controls have a maximum frequency response in the range of 0 - 10 cycles per second. If any of the helicopter drive system modes have natural frequencies in this same range, the possibility exists of the control system and drive system coupling together in an unstable divergent motion. This possibility did occur on the XH-40 (HU-1) as reported in References 3 and 4. It was impractical to change the dynamic characteristics of either the helicopter or the gas generator, so the fuel control itself was modified until the instability was removed.

The other helicopters discussed in References 2, 6, 7, and 8 do not mention any drive system instability. Since these helicopters all had lag hinges with dampers, it is concluded that no drive system - fuel control

instability occurred either because the lag hinge raised the lowest drive system frequencies above the fuel control frequency, or the damping in the lag hinge damper was sufficient to prevent instability.

The hot cycle helicopter has a rotor which is rigid in the chordwise direction as is the HU-1, and at first thought, it might be felt necessary to check the hot cycle rotor for the torsional instability that the HU-1 had. But the hot cycle rotor does not have a drive shaft between its rotor and "power turbine" as does the HU-1. (The rotor itself functions as the the power turbine.) In the simplest terms, torsional frequency is proportional to stiffness/inertia, and the hot cycle drive system has essentially infinite stiffness between rotor and "turbine". Therefore, the hot cycle rotor will not be subjected to torsional oscillation of the HU-1 type. The rotor is, however, subjected to a load torque due to aerodynamic load on the blades. If the external aerodynamic torque has any vibratory component with a frequency equal to that of the fuel control, the possibility of torsional oscillation exists. These oscillations would be limited only by blade damping. Aerodynamic oscillation will occur only at multiples of blade frequency, such as one per rev for an unbalanced blade, or at three per rev for the three-bladed hot cycle rotor.

It therefore appeared that a check should be made of the response of the rotor-gas generator-fuel control combination when subjected to vibratory aerodynamic torques in the neighborhood of one per rev and three per rev. These are the only regions where any torsional instability might exist for the jet-driven hot cycle rotor.

3.1.6 Control and Synchronization of a Two Engine Installation

References 7 and 8 include brief descriptions of the control arrangement of two different two-turbine helicopters. Both helicopters, the YHC-1A and YHC-1B, are products of Vertol and show a certain similarity. The engines however, come from General Electric (T58) and Lycoming (T55), respectively.

In each helicopter, there is a "condition" lever for each engine, with definite positions such as: "OFF", "START", GROUND IDLE", and for the YHC-1A, "FLY". These condition levers bring the gas generator from rest up to the minimum speed at which the power turbine can be controlled. The power turbines of both engines are joined by over-running clutches to a common gear box; therefore, both power turbines will operate at the same speed when power is applied. However, because of differences in component efficiency of the two engines, they can run with different gas

generator speeds when producing the same torque at the same power turbine speed. They can also produce unequal torque at the same gas generator speed and power turbine speed because there is no automatic load sharing device. In order to produce equal torques, the pilot would have to trim one engine against another by matching torquemeter readings while changing one engine gas generator speed relative to the other.

The torque matching mentioned here, as well as selection of the governed power turbine (and main rotor) speed, is accomplished on these helicopters by two power turbine selector (beep trim) switches mounted on the top of the collective stick. The left beep trim switch actuates both engines simultaneously, and was discussed earlier in Paragraph 3. 1. 4. The right switch controls only the number two engine to match the power output or gas generator speed of the number one engine.

As pointed out in Paragraph 3. 1. 4, the Sikorsky HSS-2 has two power turbine selector levers on the overhead panel, but their function is the same as the two switches mounted on the collective stick in the Vertol helicopters. An excellent time history of a torque matching operation on an earlier Sikorsky two turbine helicopter is given in Reference 2.

It should be noted here that while the hot cycle-T64 system can probably use either the Vertol or Sikorsky cockpit control lever configurations, some parameter other than engine torque will be used to match engine outputs because the hot cycle has no shaft drive system on which to measure and then match torques. The rotor is, of course, the power turbine for both engines, and a speed signal from it will go equally to both fuel controls, just as in the case of the turboshaft engines. But it will probably be necessary to use a quantity such as engine discharge total pressure as the most important item to match between engines of somewhat different thermodynamic characteristics. A test program will be proposed at the end of this report to check, among other things, the possibility of matching discharge total pressures as the best way of operating two gas generators through one hot cycle rotor.

### 3. 1. 7 Emergency Fuel Control

The General Electric T58 engine is equipped with an emergency fuel control which, in effect, bypasses the engine fuel control and permits the pilot to meter fuel directly into the engine with a throttle as in the case of older piston engine helicopters. This system is used only in emergencies and requires the pilot to monitor turbine temperature and speed very

carefully. This system is used on the single engine Sikorsky S-62A (Reference 6), and also on the Sikorsky HSS-2. However, Vertol does not use this system on their YHC-1A which has two T58 engines. (Reference 7). Perhaps more operational time on twin-turbine helicopters will disclose whether or not this emergency fuel control provision is necessary. The T64 fuel control at present does not incorporate any emergency bypass provisions, but undoubtedly, some type of emergency system could be provided if it were felt necessary.

3.1.8 <u>Review of the Preliminary Investigation</u>

The preceding review of contemporary free turbine turboshaft helicopters has established the following:

a. All free turbine engines investigated here were droop stabilized.

b. Droop stabilized governors produce static rpm droop of up to 10%; this can be corrected to 2 - 3% with a properly fashioned cam. Test agencies recommend static droop be reduced to 1%.

c. Dynamic or transient speed errors of up to 6% are currently experienced. A maximum dynamic droop of 2% is recommended and may be obtained by increasing the control gain; the system may, however, go unstable with too much gain.

d. Electrical "beep" circuits are sometimes used either to correct the residual static droop of Item 3.1.2, or to synchronize two engines as discussed in Paragraph 3.1.6. The actuation rates of circuits tested so far are too slow and should be increased to 4% change in three seconds or less. A manual system may be used to correct the residual droop but the pilot will probably have to take his hand off the collective stick in a two engine installation.

e. The drive systems of gear driven helicopters may be in resonance with the fuel control natural frequency, resulting in unstable torsional oscillations. The hot cycle system has no flexibility between the rotor and the "power turbine" and thus cannot experience this resonance. But vibratory aerodynamic loads may excite the rotor at the fuel control

frequency, producing large amplitude oscillations; a frequency response check should be made for a hot cycle rotor from one per rev up to three per rev for a three-bladed rotor to be sure the system has enough damping to keep the amplitude of oscillation to acceptable values.

f. Two turboshaft engines can be geared to one rotor with no trouble and operated with different quality engines; the engines can be trimmed to produce equal torque. The hot cycle will also operate with two engines of different quality, but some parameter such as discharge total pressure will have to be used to match the two engines against each other, rather than torque.

g. Operational experience may show the need for, and feasibility of, an emergency fuel control bypassing the main engine fuel control.

The items discussed here raised an issue which could not be answered by drawing analogies from installations which are only slightly similar. It became obvious that it was necessary to investigate the matching of the actual hot cycle rotor to the T64 engine in order to determine static or dynamic droop, etc. A complete dynamic analysis of the hot cycle rotor with two T64 gas generators was conducted, with the aid of engine and control information supplied by General Electric. The case of one engine inoperative was also studied. This dynamic analysis is reported in the next section.

## 3.2 ANALYTICAL GOVERNING STUDY OF HOT CYCLE - T64 SYSTEM

The classical method of performing a dynamic analysis of an engine-rotor control study is to derive equations of motion for all components in the complete system, over a large range of power levels. These equations of motion can involve from three to perhaps ten or eleven degrees of freedom. Each component is characterized by a "gain" and a "time constant". An example of this classical approach is outlined in Reference 3, which includes a block diagram of the system studied there, as well as the equations of motion. It is pointed out in Reference 3 that the engine manufacturer develops the equations of motion for the fuel control, gas generator, and power turbine, and determines the gains and time constants for the components under his control. The airframe manufacturer derives the equations of motion and physical data for the drive system and rotor load

functions. These combined equations are then solved in any convenient manner to determine stability, amount of damping, frequency response, and transient response to load disturbances. It has been the practice to linearize the equations and solve them on analog computers, as was the case discussed in Reference 3.

It was determined that the dynamics of the hot cycle - T64 gas generator system could be handled in the same way as a gear driven helicopter when allowance was made for the fact that the hot cycle rotor is the power turbine, turning at 243 rpm instead of 13,600 rpm, as does the power turbine of the T64 turboshaft engine. In addition, the jet drive derives its torque in a somewhat different fashion than does the conventional power turbine. With the exception of these points, which only effect the torque derivatives of the power turbine, all other engine data can be used for hot cycle jet rotor application exactly as derived by the engine manufacturer for studying the conventional gear driven rotor.

### 3.2.1 Description of General Electric T64 Turboshaft Fuel Control

A description of the fuel control for the T64 gas generator that will be used to power the hot cycle rotor is contained in Reference 9. It should be noted that the basic engine model will be the YT64-GE-6 turboshaft. Even though the power turbine will be removed from those engines to obtain gas generators, the hot cycle rotor will replace the original power turbine as the item whose speed is governed by the basic fuel control. Therefore, the discussion in Reference 9 is pertinent.

The engine control system consists of two major assemblies; a hyro-mechanical fuel control and an electrical temperature and overspeed control.[a] The fuel control controls engine power and schedules the compressor stator vane actuators. In the YT64-GE-6 installations, the fuel control also governs the free turbine speed. The electrical control limits turbine temperature and power turbine overspeed.

The complete functions of the control system are as follows:

a. Maintain free turbine (hot cycle rotor) speed at the selected level.

b. Automatically vary gas generator power to match output loading.

c. Protect engine against overspeed.

d. Protect engine against overtemperature.

e. Provide adequate acceleration and deceleration within the limits of stall and loss of combustion.

f. Provide automatic engine starting.

To perform these functions, the control monitors the following engine parameters:

(1) Power control shaft linkage position

(2) Load signal shaft linkage position

(3) Free turbine (hot cycle rotor) speed

(4) Gas generator speed

(5) Free turbine inlet temperature

(6) Compressor discharge pressure

(7) Stator vane actuator position

(8) Engine inlet temperature

The engine control system employs a cockpit engine control lever and a signal from the collective pitch lever to actuate the two control setting shafts of the fuel control, (1) the power control shaft, and (2) the load signal shaft. Engine output speed (and hot cycle rotor speed) is regulated by the position of the power control shaft. Engine output power is selected indirectly by the helicopter controls by means of increasing or decreasing the load on the engine (e.g., rotor collective pitch). The free turbine (hot cycle rotor) governor controls the gas generator speed governor to match engine power to load power. Engine starting and shutdown is also accomplished by the power control shaft.

The load signal shaft is connected by airframe supplied linkage to collective pitch (or other indication of engine load) to provide anticipation of load transients and reduces the power change required from the free turbine (hot cycle rotor) speed control. The airframe - supplied linkage must provide an indication of load in accordance with a $0^{\circ}$ -minimum load,

90° -maximum load, schedule. In helicopter applications, the maximum collective pitch angle will be coordinated with a load signal shaft at, or close to, 90° as determined by an optimization of control performance with respect to rotor characteristics and the airframe flight envelope.

It should be noted here that the load signal shaft is the item on the engine which is actuated to provide "droop" elimination as recommended in Paragraphs 3. 1. 2 and 3. 1. 8 above. Other engines, such as the T53, T55, and T58, connect the collective pitch stick through suitable linkage to the power turbine speed selector, and reset that lever as a function of load. While this apparently could be done on the T64, General Electric has pointed out to this Company that such an approach would not be desirable on the T64. The special load signal shaft is to be used for sudden large signals into the engine, rather than the power control shaft, which could be reset just as easily by the collective stick. The General Electric comment was that the basic stability of the governor was involved. Since the engine is built to operate in this fashion and will operate in the corrective fashion recommended in Paragraphs 3. 1. 2 and 3. 1. 8, the hot cycle control system discussed in this report will assume that collective pitch is properly coordinated to the load signal shaft for droop elimination.

When the power control shaft is in the normal operating range, a wide range of power turbine (or hot cycle rotor) speeds is possible. A linear schedule of rotor speed from 85% (nominally) to 100% speed is provided, with 100% speed corresponding to any predetermined rotor speed between 212 and 304 rpm, if the reference rotor speed is 243 rpm for 13, 600 power turbine rpm. If it is desired, the 243 rotor rpm can be referenced to any power turbine speed between 12, 000 and 17, 000 rpm, thus affecting the amount of rotor rpm reset the power control shaft will produce above or below 243 rpm. If, for instance, it is desired to reduce rotor rpm to 70% of 243, the reference power turbine speed would be 17, 000, and moving the power control shaft to the minimum governed position would produce 172 rotor rpm, or 70% of maximum. Whatever the desired rotor rpm in the normal operating range, the fuel control automatically varies engine power between Idle and Military to match load power at the selected power turbine speed. Whenever the load changes, there will be (1) an immediate power change if the collective pitch was moved as part of the load change, and (2) subsequent power correction by the rotor speed control as a function of speed error.

At pointed out above, the airframe requirements for installation of the T64 gas generator include linkage from the cockpit for the power control shaft and the load signal shaft. A further airframe

requirement is a flexible drive shaft connection between the airframe and the engine control. This drive shaft must feed a rotor speed signal to the engine control. This flexible shaft has several detailed requirements listed on page 11-a of Reference 9. The shaft turns at a reduced speed of 1/3.782 relative to the original power turbine. Since the hot cycle rotor will operate at 243 rpm compared to the nominal power turbine speed of 13,600 rpm, one special airframe requirement will be a gear box that will convert an rpm signal at 243 rpm to a fictitious signal at $\frac{13,600}{243 \times 3.782}$ or 14.78 times faster than rotor speed.

### 3.2.2 Equations of Motion of Rotor, Gas Generator and Fuel Control

Equations of motion and a block diagram for the T64 gas generator and fuel control were obtained from General Electric, together with the appropriate gains and time constants for each component. These equations, plus the rotor load equation, are given below, using rotor speed as the governed item rather than power turbine speed. This transfer to rotor speed $\Delta N_R$ requires that the coefficient of the $\Delta N_f$ term as supplied by General Electric be increased by 13,600/243, or 56 times. In addition, a term which corresponded to the inertia of the power turbine was deleted because this item is removed from the engine and its function and inertia are replaced by the rotor. Figure 3-1 is a slightly modified version of the General Electric block diagram and it should be referred to for easy understanding of the equations of motion. Figure 3-1 refers to rotor speed, not power turbine speed, and it includes the rotor transfer functions. Otherwise it is the same as the General Electric block diagram. The following variables are used:

(Note: All variables are increments from initial values before a disturbance)

| | | |
|---|---|---|
| $\Delta R$ | Fuel flow/compressor discharge pressure $W/P_3$ | lb/hr/psi |
| $\Delta W$ | Fuel flow | lb/hr |
| $\Delta P$ | Compressor discharge pressure | psi |
| $\Delta \%N_g$ | % change of gas generator speed | |
| $\Delta Q_G$ | Gas torque produced at rotor tip at rotor rpm | ft-lb |
| $\Delta Q_A$ | Aerodynamic torque on rotor | ft-lb |

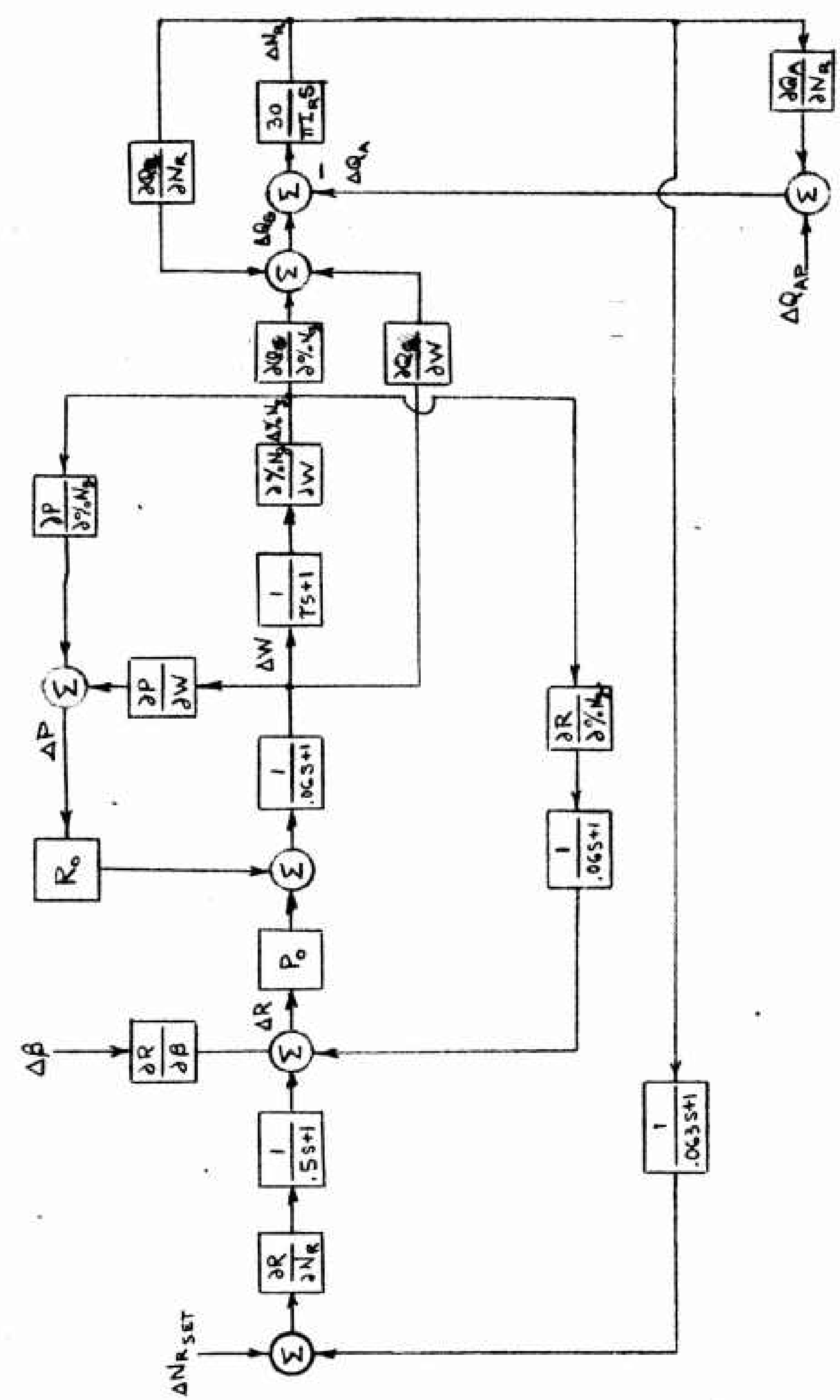


Figure 3-1. Schematic Diagram of Hot Cycle Rotor -- T64 Gas Generator Control System

| | | |
|---|---|---|
| $Q_{AP}$ | Step change of aerodynamic torque | ft-lb |
| $Q_{AA}$ | Vibratory component of aerodynamic torque | ft-lb |
| $N_{f_o}$ | Reference power turbine speed at 243 rotor rpm (taken as 13,600 rpm in all cases) | rpm |
| $\Delta\beta$ | Change in load signal shaft position due to change of collective pitch<br>(Note: 100% change in load from 0 to 100% $= 90^\circ \Delta\beta$ ) | degrees |
| $\Delta N_R$ | Change of rotor speed | rpm |
| $\Delta N_{R_{Set}}$ | Change of reference rotor speed from 243 rpm (to reduce droop when load signal shaft is not removed) | rpm |
| S | La Place operator $= \frac{d}{dt}$ | |

(Eq. 3-1)
$$\Delta R = \frac{dR}{d\%N_g} \frac{1}{.063S + 1} \Delta\%N_g - \frac{.537}{.5S + 1} \left(\frac{N_{f_o}}{13,600}\right)^{.77} \times \left(\frac{1}{.063S + 1} \Delta N_R + \Delta N_{R_{Set}}\right) + \frac{\partial R}{\partial \beta} \Delta\beta$$

(Eq. 3-2)
$$\Delta W = \frac{1}{.06S + 1} (P_o \Delta R + R_o \Delta P)$$

(Eq. 3-3)
$$\Delta P = \frac{\partial P}{\partial W} \Delta W + \frac{\partial P}{\partial \%N_g} \Delta\%N_g$$

(Eq. 3-4)
$$\Delta\%N_g = \frac{1}{\tau S + 1} \frac{d\%N_g}{dW} \Delta W$$

(Eq. 3-5) $$\Delta Q_G = \frac{\partial Q_G}{\partial W} \Delta W + \frac{\partial Q_G}{\partial \% N_g} \Delta \% N_g + \frac{\partial Q_G}{\partial N_R} \Delta N_R$$

(Eq. 3-6) $$\Delta Q_G - \Delta Q_A = \frac{2\pi}{60} I_R S \Delta N_R$$

(Eq. 3-7) $$\Delta Q_A = Q_{AP} + Q_{AA} \text{ SIN } \omega t + \frac{\partial Q_A}{\partial N_R} \Delta N_R$$

All of the engine dynamic characteristics, gains, and time constants which are included in these equations of motion and are shown in the block diagram of Figure 3-1 are given as a function of power level in Table 3-1. It should be noted that certain derivatives, such as

$$\partial Q_G/\partial N_R, \quad \partial Q_G/\partial W \quad \text{and} \quad \partial Q_G/\partial \% N_g,$$

are given relative to rotor speed, rather than the much higher equivalent power turbine speed. Using information supplied by General Electric concerning mass flow, temperature, and pressure, the differential torques produced by changes of fuel flow, gas generator speed, and rotor tip speed, are computed for a design rotor tip speed of 700 ft/second. These derivatives have the same sign as the usual turboshaft derivatives, but are much larger in magnitude, reflecting the higher torque and change of torque associated with the large, slow turning rotor.

### 3.2.3 Solution of Equations on an IBM 7090 Digital Computer

Although it is quite common to solve systems of linear differential equations such as those in the preceeding section on electric analog computers, there is no unique increase in accuracy obtained on an analog over solutions obtained on a digital computer. It is true that more engineers are familiar with solving such equations on analog computers, but a digital computer, properly used, can provide more accurate answers, and if only a relatively few cases are involved, digital solutions can actually be cheaper than analog solutions. In addition, for practically no difference in cost, a graphical time history of the variables can be obtained for quick scanning. These graphical print-outs are obtained in addition to the usual digital numerical printer sheets, so no accuracy is lost when more complete study of some point is desired.

## TABLE 3-1

## T64 ENGINE DYNAMICS AND ROTOR DERIVATIVES

| %Mil. HP | Flight Idle | 25% of Mil. | 50% of Mil. | 75% of Mil. | Military | Units |
|---|---|---|---|---|---|---|
| 1. $\frac{\partial P}{\partial W}$ | 0.029 | 0.028 | 0.028 | 0.028 | 0.027 | $\frac{\text{p.s.i.}}{\text{lb/hr}}$ |
| 2. $\frac{\partial P}{\partial \% N_g}$ | 3.40 | 5.25 | 6.10 | 5.40 | 4.05 | $\frac{\text{p.s.i.}}{\text{\%RPM}}$ |
| 3. $\frac{\partial R}{\partial \% N_g}$ | -0.653 | -0.694 | -0.721 | -0.743 | -0.768 | $\frac{\text{lb/hr/p.s.i.}}{\text{\%RPM}}$ |
| 4. $P_o$ | 65.0 | 95.0 | 122.5 | 144.5 | 168.0 | p.s.i. |
| 5. $R_o$ | 6.0 | 6.53 | 7.27 | 7.75 | 8.33 | $\frac{\text{lb/hr}}{\text{p.s.i.}}$ |
| 6. $\frac{d\% N_g}{dW}$ | 0.0477 | 0.0161 | 0.0116 | 0.0115 | 0.0115 | $\frac{\text{\%RPM}}{\text{lb/hr}}$ |
| 7. $\frac{\partial Q_G}{\partial W}$ | 17.0 | 16.0 | 13.22 | 11.30 | 10.28 | $\frac{\text{ft lb}}{\text{lb/hr}}$ |
| 8. $\frac{\partial Q_G}{\partial \% N_g}$ | 590.0 | 980.0 | 1610.0 | 1380.0 | 1292.0 | $\frac{\text{ft lb}}{\text{\%RPM}}$ |
| 9. $\frac{\partial Q_G}{\partial N_R}$ | -13.5 | -27.0 | -37.0 | -45.0 | -51.5 | $\frac{\text{ft lb}}{\text{RPM}}$ |
| 10. $\frac{\partial Q_A}{\partial N_R}$ | ← $\frac{2Q_o}{N_{R_o}^2}(N_R)$ → | | | | | $\frac{\text{ft lb}}{\text{RPM}}$ |
| 11. $N_{\xi_o}$ | Equivalent governed known turbine speed = 13600 | | | | | RPM |
| 12. $\tau$ | 1.06 | 0.36 | 0.26 | 0.26 | 0.259 | seconds |
| 13. $\frac{\partial R}{\partial \beta}$ | 0.25 | 0.16 | 0.11 | 0.13 | 0.18 | $\frac{\text{lb/hr/p.s.i.}}{\text{degree}}$ |

The method of solution of the equations of motion is extremely simple, using available FORTRAN routines. Each of the equations (3-1) to (3-7) is first solved for the highest time derivative of a variable in it. Using equation (3-4), for example,

$$\Delta \% N_g = \frac{1}{\tau S + 1} \left( \frac{d\% N_g}{dW} \right) \Delta W$$

$$\tau S \ \Delta \% N_g + \Delta \% N_g = \left( \frac{d\% N_g}{dW} \right) \Delta W$$

$$\tau \Delta \% \dot{N}_g + \Delta \% N_g = \left( \frac{d\% N_g}{dW} \right) \Delta W$$

$$\Delta \% \dot{N}_g = \frac{\dfrac{d\% N_g}{dW} \Delta W - \Delta \% N_g}{\tau}$$

All of the equations are transformed in this fashion. Then an available FORTRAN routine for numerical solution of differential equations is used. This FORTRAN routine uses the Adams method, which is described in Reference 10. The Adams method is a procedure wherein, to solve differential equations numerically, the derivative of a function is replaced by a polynomial and the polynomial is integrated over an interval.

An important factor in using the Adams method is that the problem time interval between steps must be small enough to follow accurately the motions of the highest frequency mode in the problem. According to Reference 11, a sufficiently small computing interval is 1/30 of the period of the highest frequency. The most practical way to check the computing interval is to make an initial guess of the time interval, solve the problem, and then repeat with a computing interval of 1/10 of the original estimate. If the answers do not change noticeably, the original time interval was satisfactory. Naturally, the longest computing interval is to be sought since computing time and therefore, cost, is reduced. In all cases investigated in this study, the longest computing interval was always sought.

### 3.2.4 Transient Response Following a Small Disturbance

The time history of rotor rpm was computed for the hot cycle rotor T64 gas generator system shown in Figure 3-1 following a step input of + 10% of maximum torque with the system initially in equilibrium at 90% of military power. This + 10% change of torque was combined with a + 9° (or 10% of full travel 90°) signal on the load signal shaft to take advantage of the special droop compensation on the T64 fuel control. This time history is shown in Figure 3-2, Curve A. It is seen that the rotor speed error reaches about -0.1% rpm in less than 0.2 seconds, reverses to +0.205% rpm at 0.5 seconds, and then decays to +0.05% rpm in about 2.0 seconds in a very heavily damped motion. (The initial rotor rpm was 243). Therefore, the maximum speed error of +0.205% for a 10% torque change would correspond, in a completely linear system, to $\frac{+0.205 \times 10}{243}$ or +2.05% dynamic error, very close to the maximum dynamic droop of 2% recommended in Section 3.1.8.

Curve A shows that the static rpm change is actually slightly positive after a positive torque change, whereas in the conventional system a positive torque change would have a negative residual rpm change. This reversal is brought about by the load signal shaft, which is apparently so strong, it can overcorrect if used on a uniform percentage basis with the torque change. If a $\Delta\beta$ signal of about +8.5° had been used for the +10% torque change instead of +9°, the static rotor rpm error would have been zero, or less than the 1% maximum recommended in Section 3.1.8. It may be concluded from these results that the load signal shaft on the T64 may be used with no change to reduce dynamic and static rpm droop to the recommended values.

It is of interest to check the rotor rpm behavior without using the load signal shaft at all. Curve B on Figure 3-2 shows a maximum speed error of -1.29% rpm at 0.8 seconds which decays to -0.92% rpm after about 3 seconds. This represents a dynamic droop of -1.29 x 10 x 100 or -12.9% and a static droop of -0.92 x 10 x 100 or -9.2%. These values are well in excess of the recommended values of Section 3.1.8 and would probably be wholly unacceptable in operation. The fact that these droops can be reduced to the recommended values, as shown in Curve A of Figure 3-2, shows that the load signal shaft certainly should be used in hot cycle rotor applications.

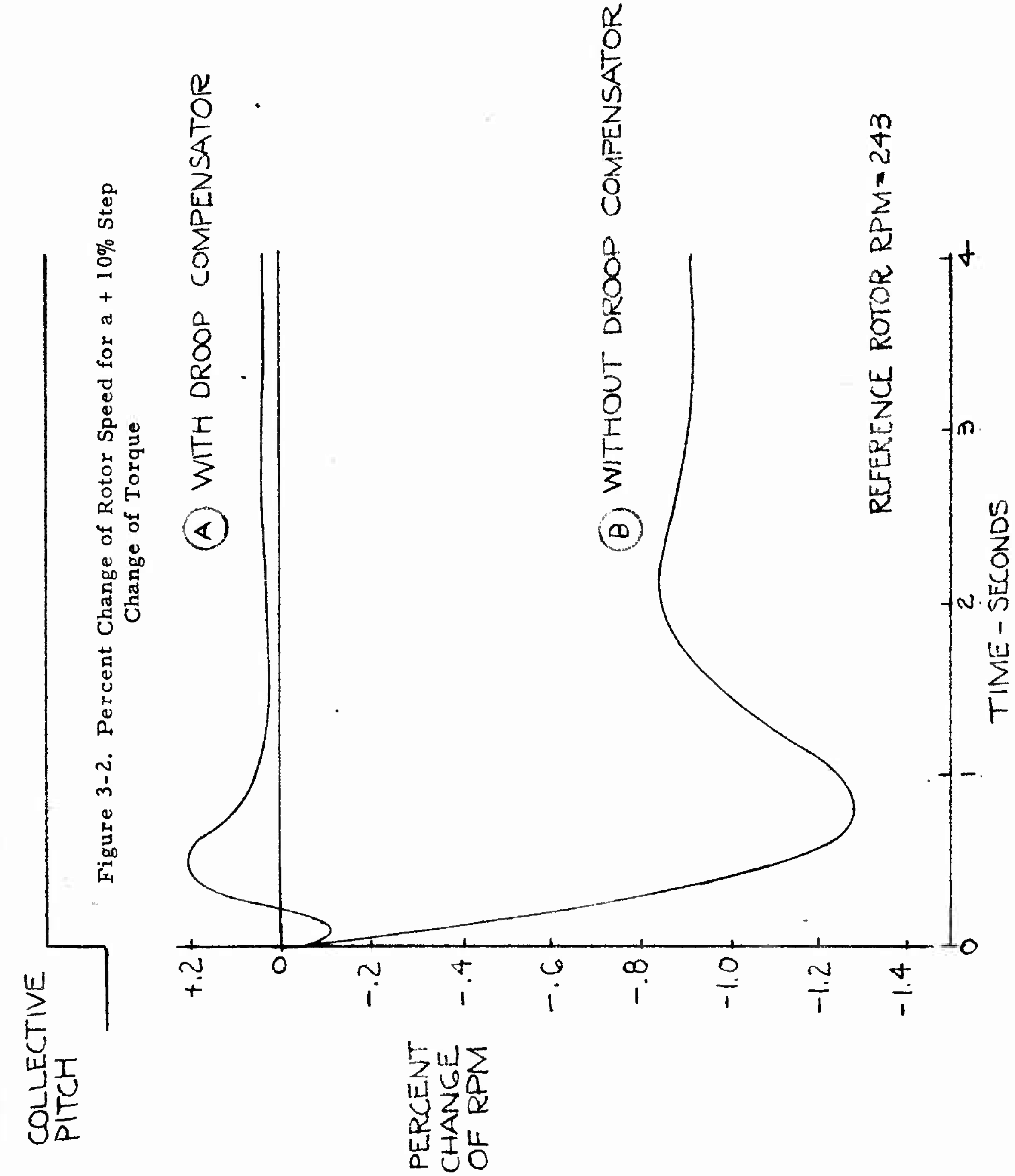


Figure 3-2. Percent Change of Rotor Speed for a + 10% Step Change of Torque

3.2.5 Frequency Response For Small Disturbances

As recommended in Section 3.1.8, the frequency response of the combined system was found for the frequency range of 0.1 cycles per second to 13 cycles per second. This represents a range from almost zero to greater than three per rev. An oscillating torque of $\pm$ 7% was used, with no $\Delta\beta$ signal. Based on the static droop of -0.92% rpm from a 10% torque change as found in the preceding paragraph, the 7% torque used here would produce a static droop of -0.64% rpm. The frequency response curve is shown in Figure 3-3, and it shows a maximum speed error of 1.11% rpm at 0.4 cps. Compared to the expected static error of -0.64% rpm, this is an amplification of 1.11/0.64, or about 1.7 times static at the resonant frequency of the governor. However, the only expected frequency inputs will be at one per rev and three per rev as marked on Figure 3-3. The rpm response for these frequencies is only 0.08% rpm and 0.003% rpm respectively, which is practically no response at all. Therefore, it may be stated that the rotor system and engine fuel control will operate well away from resonance with each other and no large amplitude oscillations will be encountered in operation.

It is of interest to note that the maximum frequency response shown in Figure 3-3 occurs at 0.4 cps, which is 2.5 seconds per cycle. It can be seen in Figure 3-2 on curve A or B that the period of the natural frequency of the system in response to a step input is exactly 2.5 seconds, showing excellent agreement between what are two different mathematical approaches to the same physical problem.

It must be pointed out that the 2.5 second period found here assumed that the gas driving the rotor immediately produced the indicated effect as soon as the engine responded to the fuel control. That is, if the fuel increased, the change in rotor torque due to fuel flow

$$\frac{\partial Q}{\partial W} \Delta W,$$

and the change in torque due to gas generator speed

$$\frac{\partial Q}{\partial \% N_g} \Delta \% N_g$$

occurred at the rotor tips as soon as the gas conditions (which would produce these changes at the rotor blade tip) existed at the back of the engine. This result would require that the rotor gases be incompressible and transmit changes instantly from the back of the engine to the rotor blade tip.

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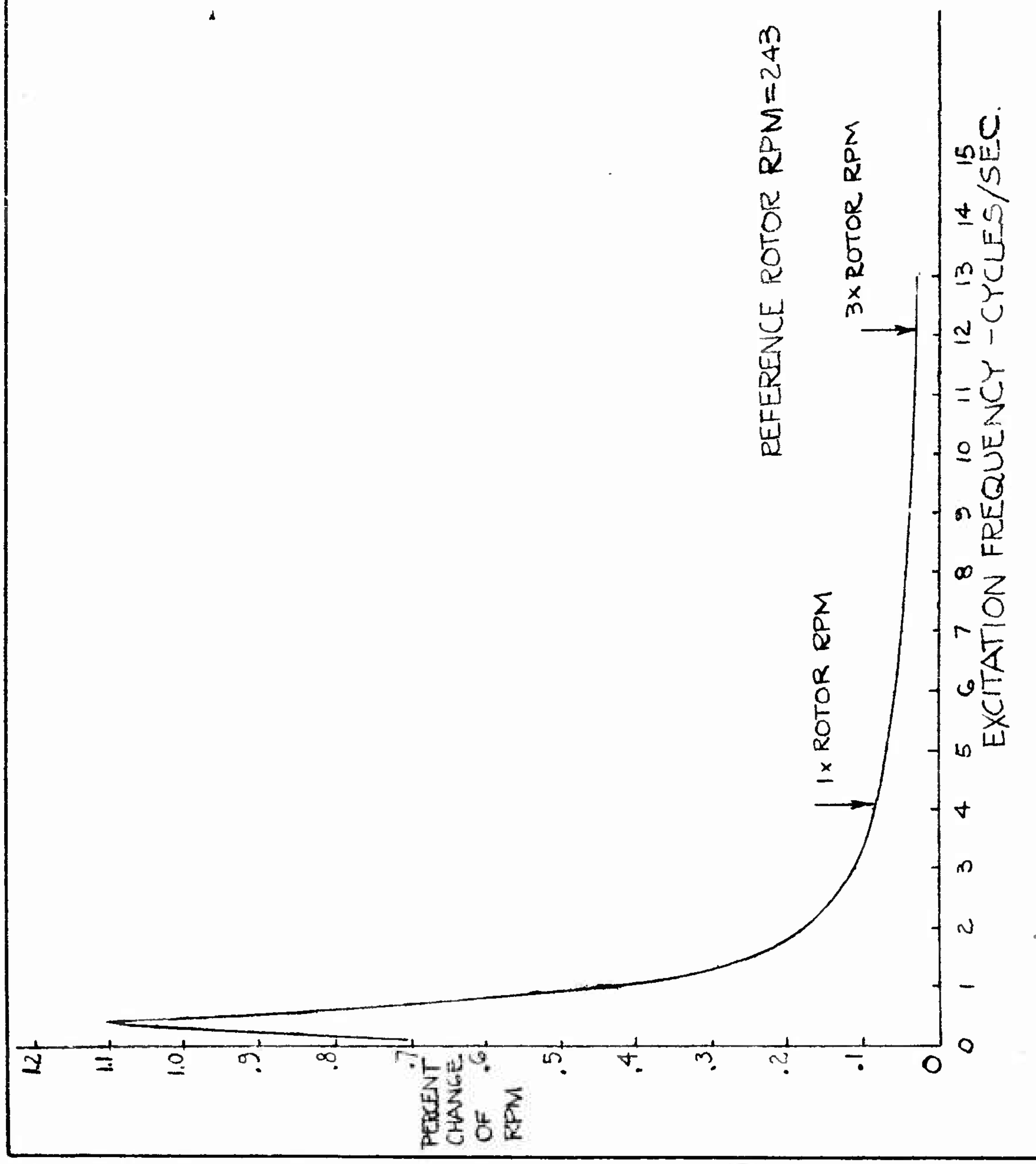


Figure 3-3. Frequency Response of System-Percent Change of Rotor RPM Vs. Excitation Frequency

Actually, the gases are compressible, and a change of gas conditions will take a finite time to be felt at the blade tip after they existed at the back of the engine. This distance is about 35 feet. The speed of sound, or the speed of a pressure wave in the gas is $49\sqrt{T}$. The idle gas static temperature is 780°F, or 1240°R. The velocity of sound corresponding to this temperature is 1720 feet per second. Consequently, a pressure pulse will take $\frac{35}{1720}$ = .02 seconds to go from the engine to the blade tips.

Note: A mass change or temperature change will move through the system at about 0.35 Mach number. This corresponds to .35 x 1720 or 600 ft/sec. Therefore, mass or temperature changes will take 35/600 = 0.06 seconds to go to the blade tips from the engine.

Since the predominant natural frequency has a period of 2.5 seconds, it is seen that the time effect of mass, pressure, or temperature changes is still well below the time response of the system and will not couple into the dynamic equations in Section 3.2.2.

### 3.2.6 Transient Response Following Failure of One Engine

Time histories were computed of the response of rotor speed, fuel flow, and gas generator speed following the failure of one engine when both engines were initially at 50% power. This condition is of interest because the remaining good engine has the capability of accelerating to full power and provide the same power that both engines did before one failed. If the good engine responds fast enough, the pilot will not have to take any rapid corrective action as is usually required after engine failure in a fixed wing airplane. The pilot, in fact, should be able to continue his flight without even changing his collective pitch stick because he will end up with the power for which the aircraft is trimmed.

Although the pilot will not have to move the collective stick because he will have the right power, at the same time he will not get any signal from the load signal shaft which is connected to the collective stick and therefore, will not move. The only signal to the fuel control to take corrective action will come from the rotor as its speed decays when one engine quits. It is therefore expected that the dynamic and static droops will approach those values found earlier in Section 3.2.4 for small disturbances without droop compensation. Whatever static droop occurs can, of course, be corrected by resetting the reference rotor speed. It is desirable to keep all corrective actions to a minimum.

In addition to not having any load signal shaft correction, the fuel control will receive such a large signal that it will very likely encounter the fuel acceleration schedule. The acceleration schedule is a program of fuel flow versus gas generator speed that restricts the fuel to values below those which will cause compressor stall or engine flame-out. If the acceleration schedule is reached, the engine will therefore not receive as much fuel as is required by a linear system such as defined by the equations in Section 3.2.2. The restriction of fuel will, of course, slow down the build up of engine power, and rotor speed decay will be higher because of the limiting action of the acceleration schedule. In addition, the time lapse before the rotor is restored to the droop rpm will be longer because of the presence of the acceleration schedule.

Figure 3-4 presents a fuel-to-run curve and an estimated acceleration schedule. The fuel is cut off at 1361 lb/hr, which produces the maximum gas temperature that General Electric representatives said the fuel control would permit.

A case was computed, starting with the engine initially at 50% power, or 860 lb/hr. fuel. Where normally the whole rotor inertia is accelerated by two engines, or half the inertia is accelerated by one engine, for the case at hand, the whole inertia is accelerated by one engine. At the same time, the good engine feels a step change of -50% of maximum torque. Because the engine will accelerate from 50% to 100% of military power, the system derivatives used in this case will be taken from Table 3-1 at 75% military. Examination of Table 3-1 shows only moderate changes in the system parameters in the 50% to 100% power region. Therefore, the 75% power derivatives should provide a reasonably good approximation.

The time histories of rotor speed, fuel flow, and gas generator speed for this one-engine-out problem are shown in Figure 3-5. It is seen that rotor speed decays to a maximum of -12.3 rpm (5%) in 1.3 seconds and then approaches the static droop speed of -9.7 rpm (-4%) which is first reached in about 6.4 seconds. Fuel flow starts to respond immediately, reaching the cut off value of $\Delta$ W = 501 lb/hr. after 1.15 seconds. The rotor rpm reversed when it did (shortly after fuel reached its peak value) because the gas torque finally exceeded the aerodynamic torque and at that instant the rotor speed started back up again. When the rotor speed error had decreased to less than the static droop error of -4% rpm at 6.4 seconds, the fuel finally came off the acceleration limit. The rotor, gas generator, and fuel flow are all at the static droop values in about 8.5 seconds. In this case, because the rotor speed droop is -4%, the pilot would probably decide to reset the rotor speed.

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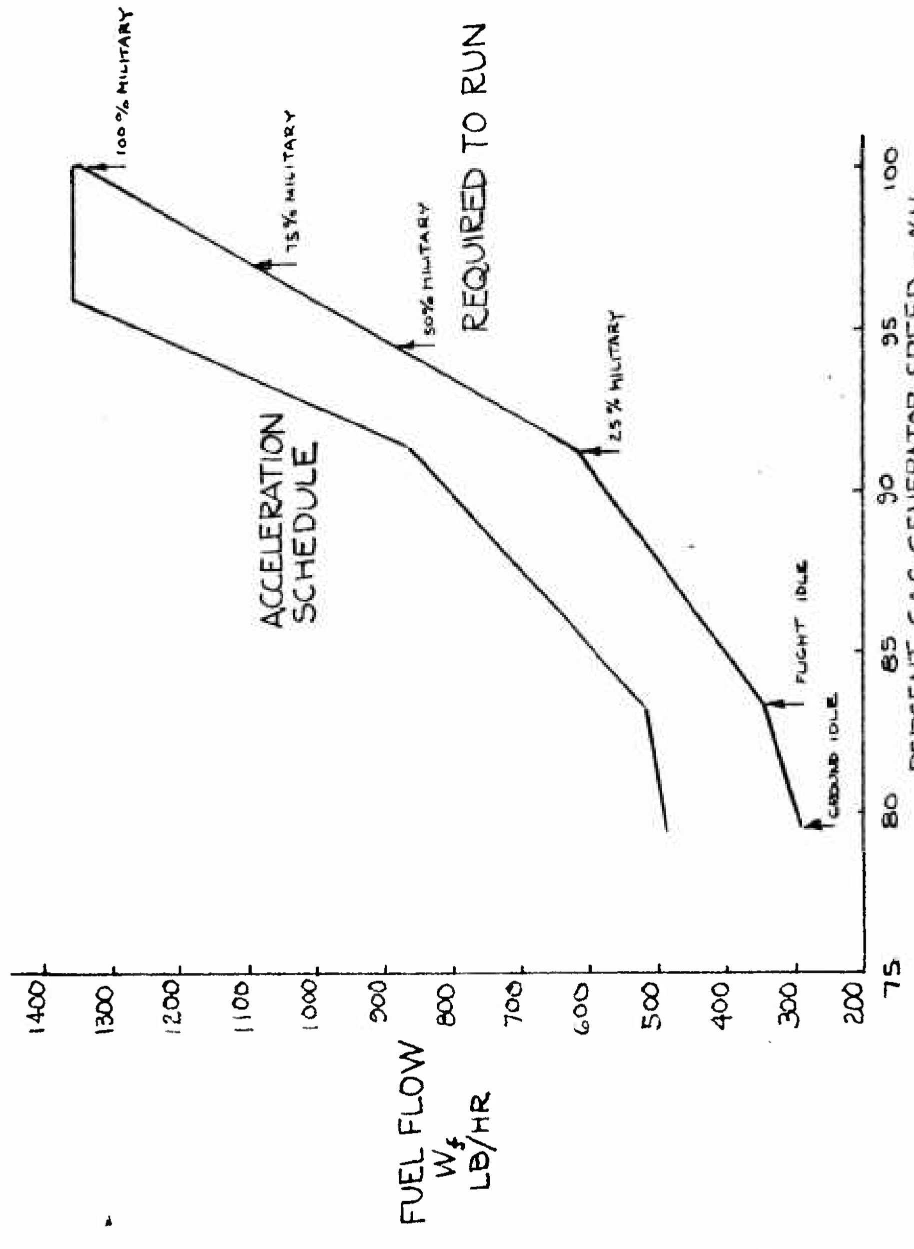



Figure 3-4. **Estimated Acceleration Fuel Schedule and Required to Run Fuel Vs. Gas Generator Speed.**

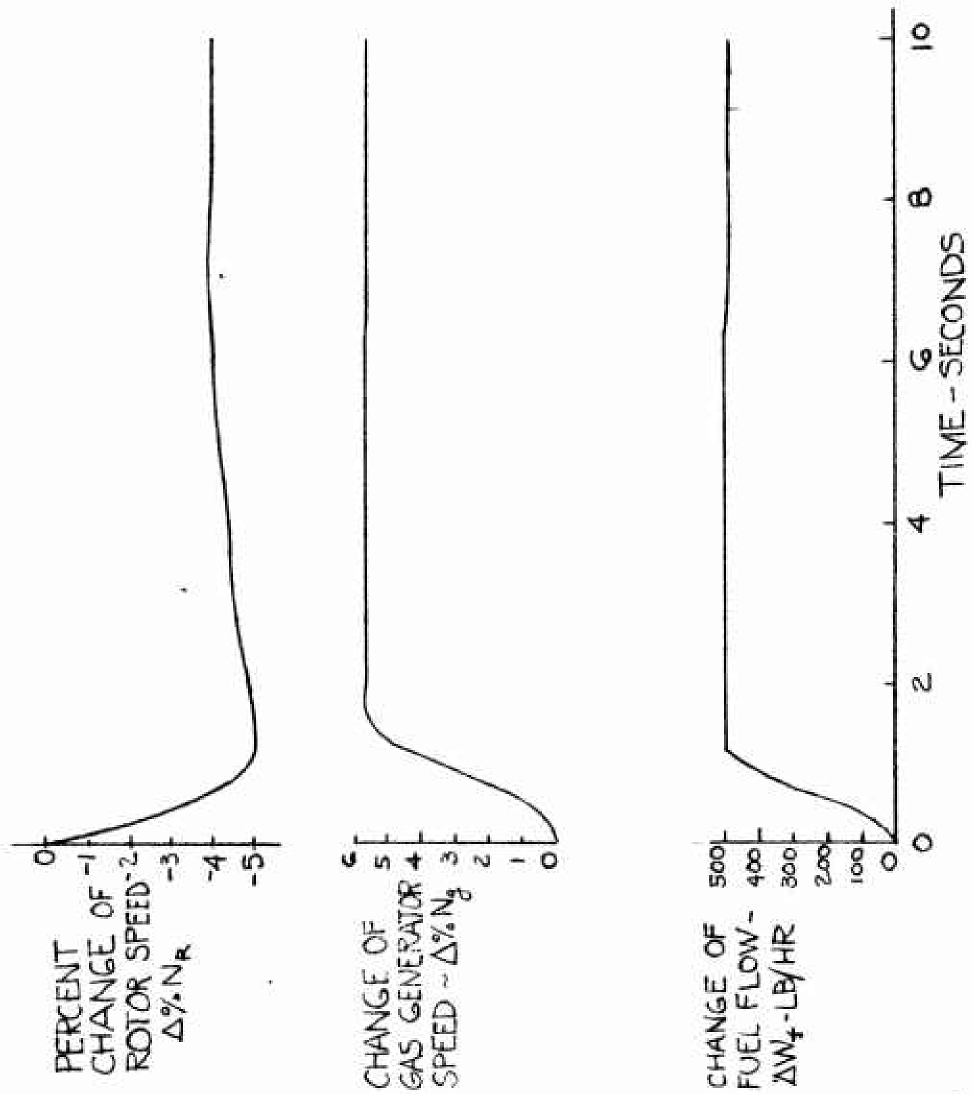



Figure 3-5. Transient Response After Failure of One Engine With Two Engines Initially at 50% Power.

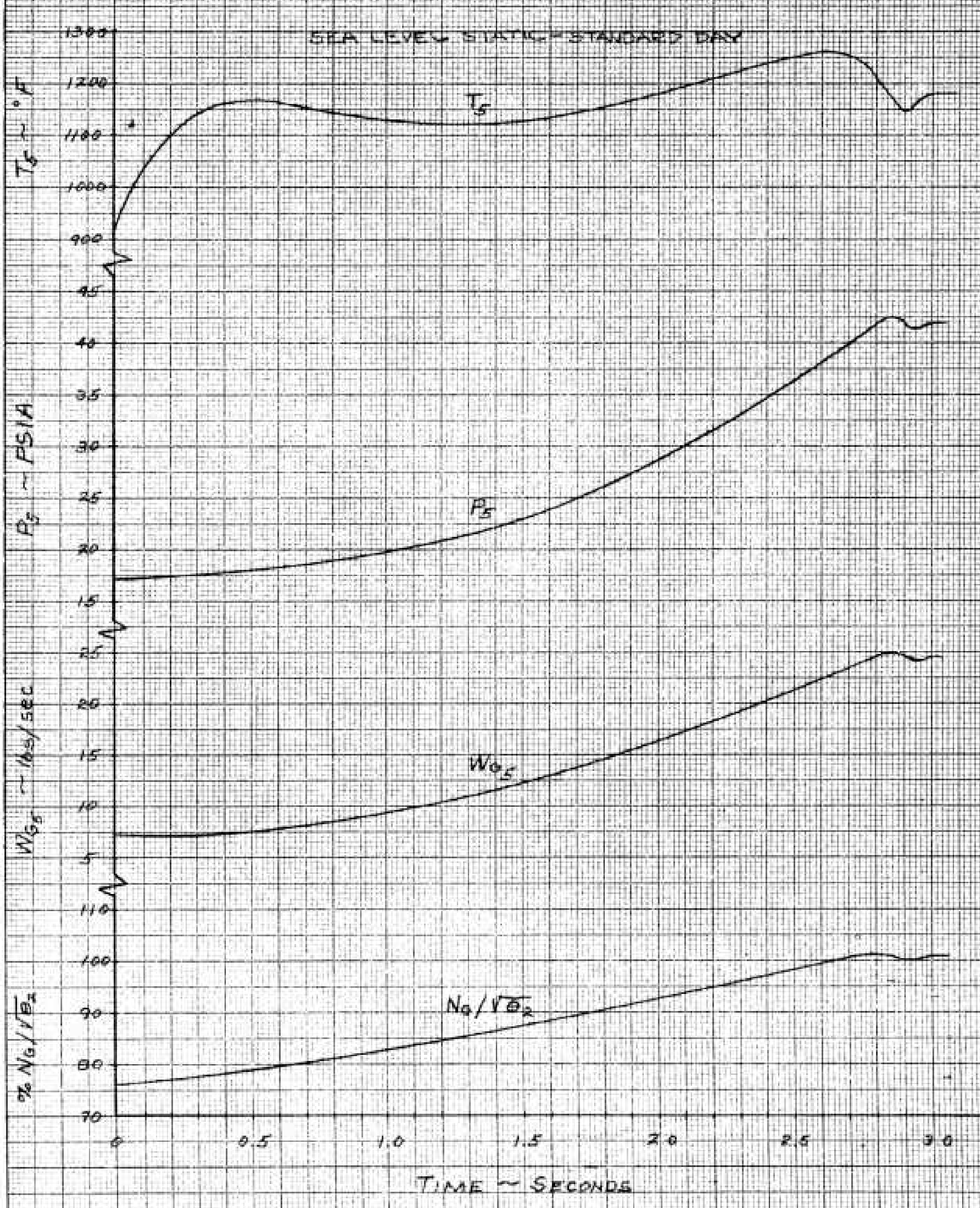
The conclusion that may be drawn from Figure 3-5 is that the remaining good engine will quickly respond to the failure of one of two engines, provide full power within about two seconds, and restore rotor rpm to the droop value in about six seconds without moving the collective pitch stick. This automatic response of the good engine in the hot cycle application is in good agreement with the case in Reference 2 which described, for a shaft-driven helicopter, single engine failure in twin engine flight. In that case, full power was achieved in two seconds and rotor speed was restored in four seconds. It therefore appears that the one engine out case will not be any different for the hot cycle case than for the conventional shaft-driven case.

### 3.2.7 Power Recovery From Practice Autorotation

The rotor speed during power recovery from a practice autorotation with two engines operating was computed. It was assumed that at the start of the recovery, the pilot had just pulled full-up collective pitch from full-down (autorotative) pitch. Therefore, the initial decelerating torque on the rotor was -100% military torque. At the same time, because the collective pitch was moved full travel, a full ($90^o$) signal would go to the load signal shaft. This large signal would demand so much fuel that the fuel acceleration schedule would be followed all the way up from idle gas generator speed. This mode of operation, according to General Electric, will invalidate the linear equations they supplied, which are supposed to apply to small power steps in the linear region only. For such large power bursts from the low end, which the information in Table 3-1, shows to be a very non-linear area, it is necessary to compute the available torque in a different manner. The procedure recommended by General Electric is to compute rotor torque using the gas condition themselves. General Electric therefore supplied Figure 3-6 to the Contractor, which is a time history of gas temperature, pressure, mass flow, and gas generator speed, all versus time, from 0 to 3.0 seconds after full throttle burst including $90^o$ motion of the load signal shaft.

The pressure and temperature were converted to jet velocity, assuming no net pressure loss between the engine and the blade nozzle, (Blade friction pressure drop assumed equal to centrifugal pressure rise.) Then with the given mass flow and rotor radius, rotor torque available was computed for 700, 600, and 500 feet per second tip speeds for every 0.1 second from 0 to 3.0 seconds. The rotor aerodynamic torque was assumed to vary with the square of the rpm; starting from the initial value of -100% military. The gas torque was, of course, almost zero at time zero, and the rotor speed was the usual initial value of 243 rpm.

Figure 3-6. Estimated Variation of T64 Gas Generator Discharge Conditions During Throttle Burst From Idle to Maximum

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The time history of rotor rpm was computed by an iterative procedure. For each time interval, starting from the beginning rpm, a final value was assumed. The gas torque was obtained, the aerodynamic torque was computed using the $(\text{rpm})^2$ rule, and the change in tip speed over the interval was computed as follows:

$$Q_G - Q_A = I\alpha = I \; \Delta \dot{N} \; \frac{2\pi}{60}$$

$V_{T_t}$ = Tip velocity at time t

$I = 3838.5 \text{ slug-ft}^2$ (one half of total rotor inertia per engine)

$$V_T = \frac{N}{243} \times 700$$

$$\dot{V}_T = 288 \; \dot{N}$$

$$\dot{N} = \frac{\dot{V}_T}{2.88}$$

$$Q_G - Q_A = 3838.5 \times \frac{2\pi}{60} \times \frac{\dot{V}_T}{2.88} = 139.8 \; \dot{V}_T$$

$$\Delta \dot{V}_T = \frac{Q_G - Q_A}{139.5}$$

$$V_T = V_{T_t} - \dot{V}_T \; \Delta t \qquad (\text{at} = 0.1 \text{ second})$$

When the computed $\Delta V_T$ was equal to the assumed $\Delta V_T$, the iteration was completed.

The results of this computation are shown in Figure 3-7. It is seen that rotor speed decays approximately 70 rpm (29%) at the end of 1.9 seconds and then starts to recover, reaching -22 rpm (-9%) at the end of 4.0 seconds. Torque available exceeds torque required after 1.9 seconds and reaches its peak value after 2.8 seconds. At the end of 4 seconds the torque available is still substantially higher than the torque required, so the speed error will decrease rapidly and reach zero error in about six seconds.

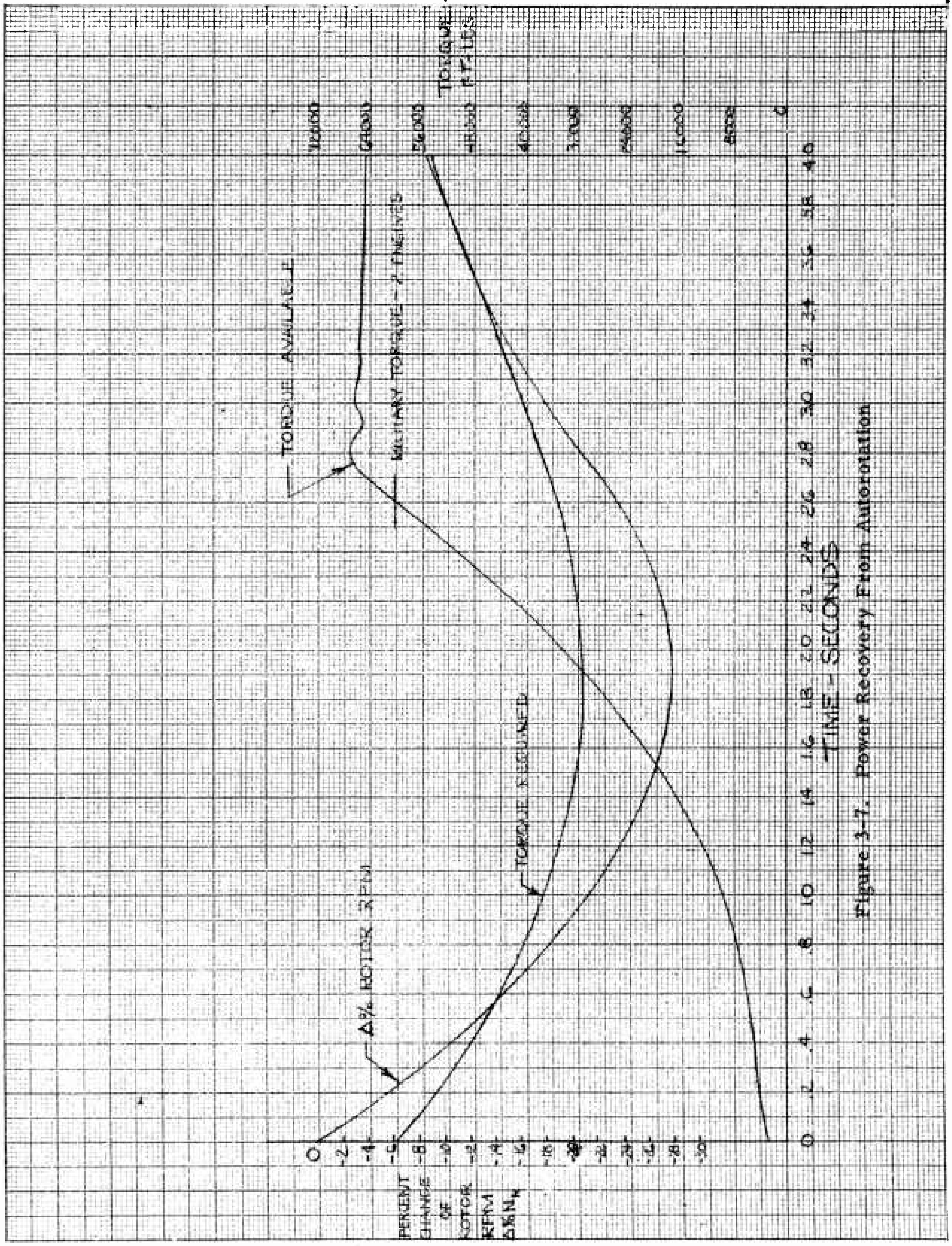



Figure 3-7. Power Recovery From Autorotation

This power recovery from autorotation will only be used in practice because normally an approach to land will be made power on. The computation here serves to point out that there is about a six second lag between application of full collective pitch and the time the rotor rpm returns back to its initial value.

This value of about six seconds to restore hot cycle rotor rpm to its original value compares very favorably to the value of about six seconds given in Reference 12 for free turbine turboshaft installations.

# SECTION 4

## PHASE II - ONE ENGINE OUT DESIGN STUDY

The objective of this phase of the study, according to Reference 1, is "to establish the requirements for selector valves and check valves to permit operation with one engine out. A preliminary design shall be made of the required valves and seals. This study shall also establish whether valve actuation shall be automatic or pilot operated."

Preliminary investigation of several hot cycle helicopter configurations has established that any two engine hot cycle helicopter will utilize a diverter valve for each engine and a blade duct valve for each blade. The exact configuration of a helicopter will determine the best valve design and location; any two engine configurations will require two diverter valves and three blade valves.

### 4.1 FUNCTIONS OF DIVERTER VALVES AND BLADE DUCT VALVES

The diverter valves discussed here will always be immediately downstream of the gas generators and will allow the following operations:

a. Direct gas overboard during engine starts rather than to rotor. If gas were directed to rotor during starts, an occasional "hot" start might damage the blade unnecessarily. Further, if ignition failed to occur temporarily, liquid fuel might accumulate in the rotor and cause serious damage if it ever later ignited in the blade. Thus, it seems safer to make all starts with diverter valves in the "overboard" position, directing gas to the rotor only when the gas generator is running properly.

b. Direct gas to the rotor during powered flight. This is the standard position of the valves for powered rotor operation. Pressure drop through the valve in this position should be a practical minimum. Leakage around doors should also be a minimum.

c. Act as check valves after one engine fails. All configurations investigated join the flow from the two engines to a common plenum at the rotor hub. Therefore, if one engine malfunctions and is shut off, the hot pressurized gases from the remaining good engine could force their way backward through the dead engine. These gases would not hurt the gas generator turbine which is designed for such temperatures. But they could damage the compressor, which is not designed to take combustion gases. Since the possibility exists that an indicated failure is a minor one, it seems reasonable to want to protect the failed engine so that it will not

become unnecessarily damaged. If the diverter valves also act as check valves, they will protect the failed engine. Also, it will be desirable to practice the one engine out case and deliberately shut off an engine. In this case, the hot gases would actually damage a good engine if check valves were not in the system.

The blade duct valves are necessary to maintain proper match between nozzle area and engine. The rotor blades are built with two ducts of approximately equal area, each with it own fixed area nozzle at the blade tip. The sum of the three tip nozzle areas is very nearly the same as the exit area for ordinary jet nozzles for two T64 gas generators. The engines behave in the helicopter installation approximately as jet engines would in a jet airplane; nozzle area is a major factor in controlling engine performance. If nozzle area is too small, an engine will surge and stall and be seriously damaged. If nozzle area is too large, the engine will operate with reduced "back pressure" and the compressor will operate at a low over-all pressure ratio. Consequently, performance will suffer seriously. For the case at hand, with two equal area nozzles on each blade tip, the effective nozzle area for one engine will double if one engine quits or shuts off and no reduction of effective nozzle area is made. By reducing the tip nozzle area in half, the remaining engine can operate up to military power.

Performance computations were made of the performance of the T64 gas generator with a 100% increase of blade tip effective nozzle area but with a sonic throat just downstream of the engine. It was found that the nozzle discharge pressure ratio was about 1.5, compared to almost 2.9 which can be obtained with standard nozzle area. At a fixed maximum gas generator speed, the mass flow will be roughly the same with 100% or 200% of standard nozzle area. Temperature will also be the same. The result of the pressure change is that the remaining engine will develop only about 52% of the power it could with proper nozzle area. Therefore, the available total power is only $52\%/2 = 26\%$ of the original power. It is not possible to cruise in level flight with only 26% of the normally available power. Approximately 50% is required. Thus, to permit worthwhile single-engine operation some way must be found to reduce the total tip nozzle area by roughly 50% to match the single engine.

Two ways of reducing effective blade tip nozzle area have been considered. The first was use of a two-position nozzle at the blade tip. The second was use of a two-position valve at the blade root to shut off one of the two blade ducts at a point just outboard from the transition from the single round duct leading from the rotor hub.

Each of these two types of valves has advantages and disadvantages. The tip valve permits use of both ducts for the gas flow from one engine. The duct Mach number in the single engine case would thus be about 1/2 that of the two engine case, and a 4% power increase can be obtained compared to the case which confines the flow from one engine to one duct. The tip location of the valve is subjected to angular accelerations of the order of 600 g's, which will be a severe design condition. On the other hand, the blade root valve would force all the gas into one duct, with somewhat higher pressure losses leading to the 4% power penalty mentioned above. Also, in order to permit use of one duct in each blade, special provisions must be made for sealing between the two ducts, and some weight penalty is involved in providing adequate structure for this condition. The location of valve at the root will reduce the angular acceleration to about 150 g's, which should be a somewhat easier design condition than for the valve located at the blade tip.

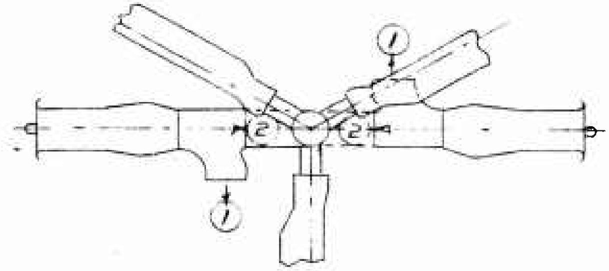
The final choice of duct valve location must be based on further study. Typical designs are discussed under Paragraph 4.4. In any case, the system dynamics would be essentially same for the valves located at either the blade root or tip.

## 4.2 GENERAL ARRANGEMENT OF GAS GENERATORS AND DIVERTER VALVES

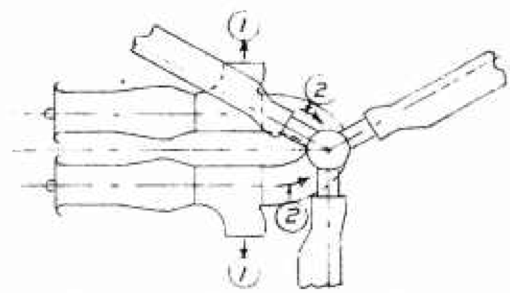
The preceding section pointed out that each helicopter configuration studied so far has had two diverter valves and the functions of those valves was outlined. Before the design of specific valves can be discussed or evaluated against each other, it is necessary to point out that the geometry of a diverter valve is influenced by the propulsion configuration of the helicopter in which it is installed. Also, in some cases, lower pressure drops between the engine and rotor will result from certain arrangements of engine and valves. At the time of writing of this report, no one helicopter configuration was a final choice among several being studied. Therefore, preliminary designs of several diverter valves were made, one or more for each of three promising engine-valve arrangements. Each of the valves would have certain size, weight, cost, pressure drop, and actuation forces. All of them will perform the functions outlined earlier. None of them will compromise the dynamic behavior of the engines as outlined in the Phase I study. No recommendation will be made in this report as to the best overall engine-diverter valve-blade valve arrangement, but a final decision will be made in later work now being started as to the best engine-valve arrangement considering all factors.

These potential arrangements of engines and diverter valves are shown in Figure 4-1. The final choice will probably be one of those shown

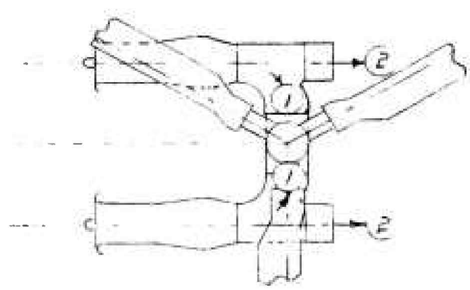
Figure 4-1. Engine and Diverter Valve Arrangements



A. FORE & AFT ENGINES,
(1.) DIVERT GAS OVERBOARD.
(2.) STRAIGHT THRU GAS TO ROTOR.



B. TWO FORWARD ENGINES.
1. DIVERT GAS OVERBOARD.
2. STRAIGHT THRU GAS TO ROTOR.



C. TWO FORWARD ENGINES.
(1.) DIVERT GAS TO ROTOR.
2. STRAIGHT THRU GAS OVERBOARD.
THIS GAS ALSO ACTS AS A PROPULSIVE JET.

ENG. & DIVERTER VALVE ARRANGEMENTS.

here. Each arrangement has one valve per engine, with gas directed overboard for starting and to the rotor for normal powered flight. The features are as follows:

### 4.2.1 Figure 4-1a

The engines are located fore and aft on the centerline of the helicopter. The diverter valves direct gas overboard in the diverted position. The gas exhausts primarily at right angles to the helicopter centerline. When the valves are turned to the straight through position, gas flows directly to the rotor.

### 4.2.2 Figure 4-1b

Both engines forward on each side of the helicopter centerline. The diverter valves direct gas overboard in the diverted position. The gas exhausts primarily at right angles to the helicopter centerline. When the valves are turned to the straight through position, gas flows directly to the rotor.

### 4.2.3 Figure 4-1c

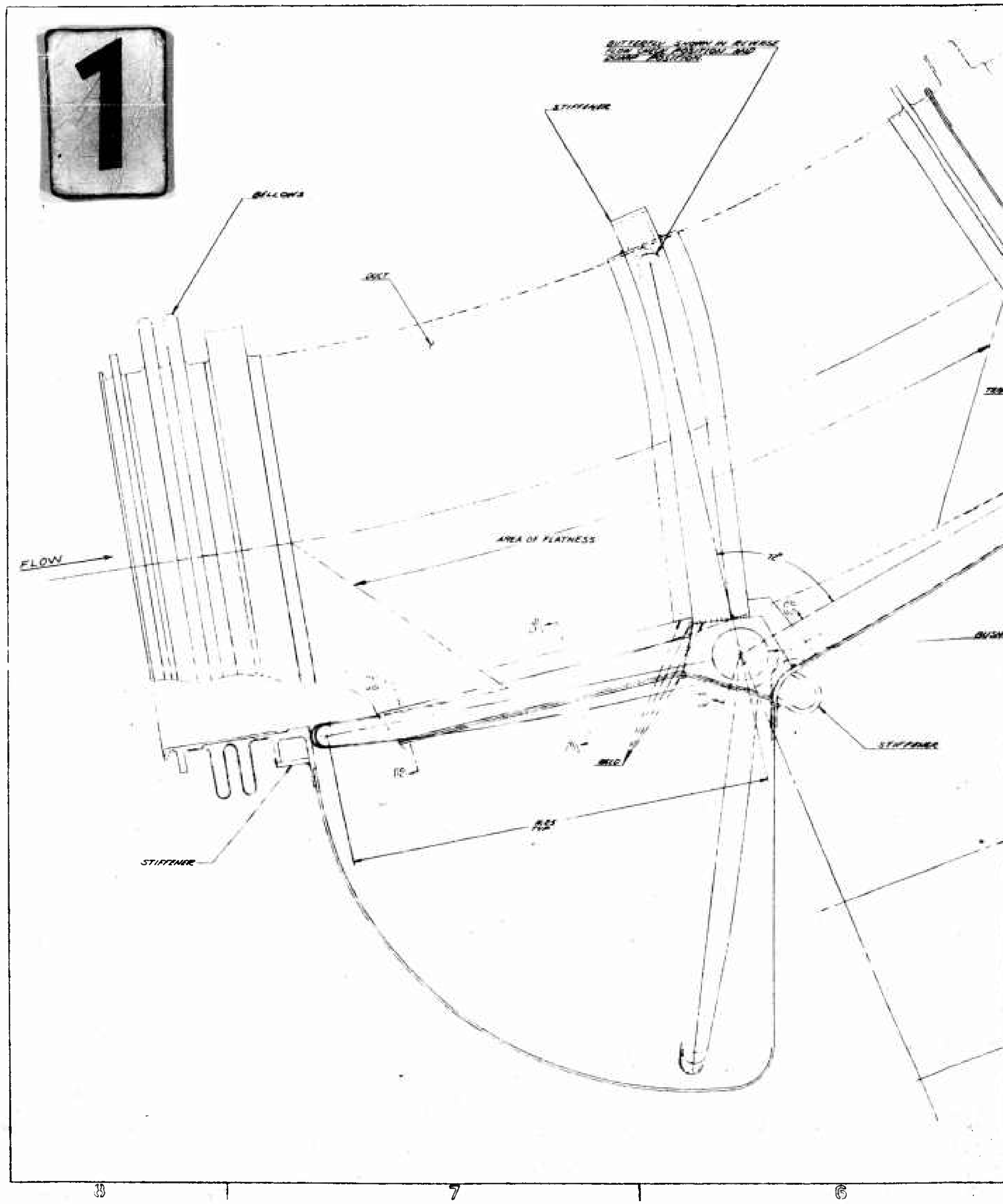
Both engines forward on each side of the helicopter centerline. The diverter valves direct gas overboard in the straight through position. The gas exhausts almost directly to the rear and can serve as a propulsion jet if desired. When the valves are turned to the diverted position, gas flows directly to the rotor.

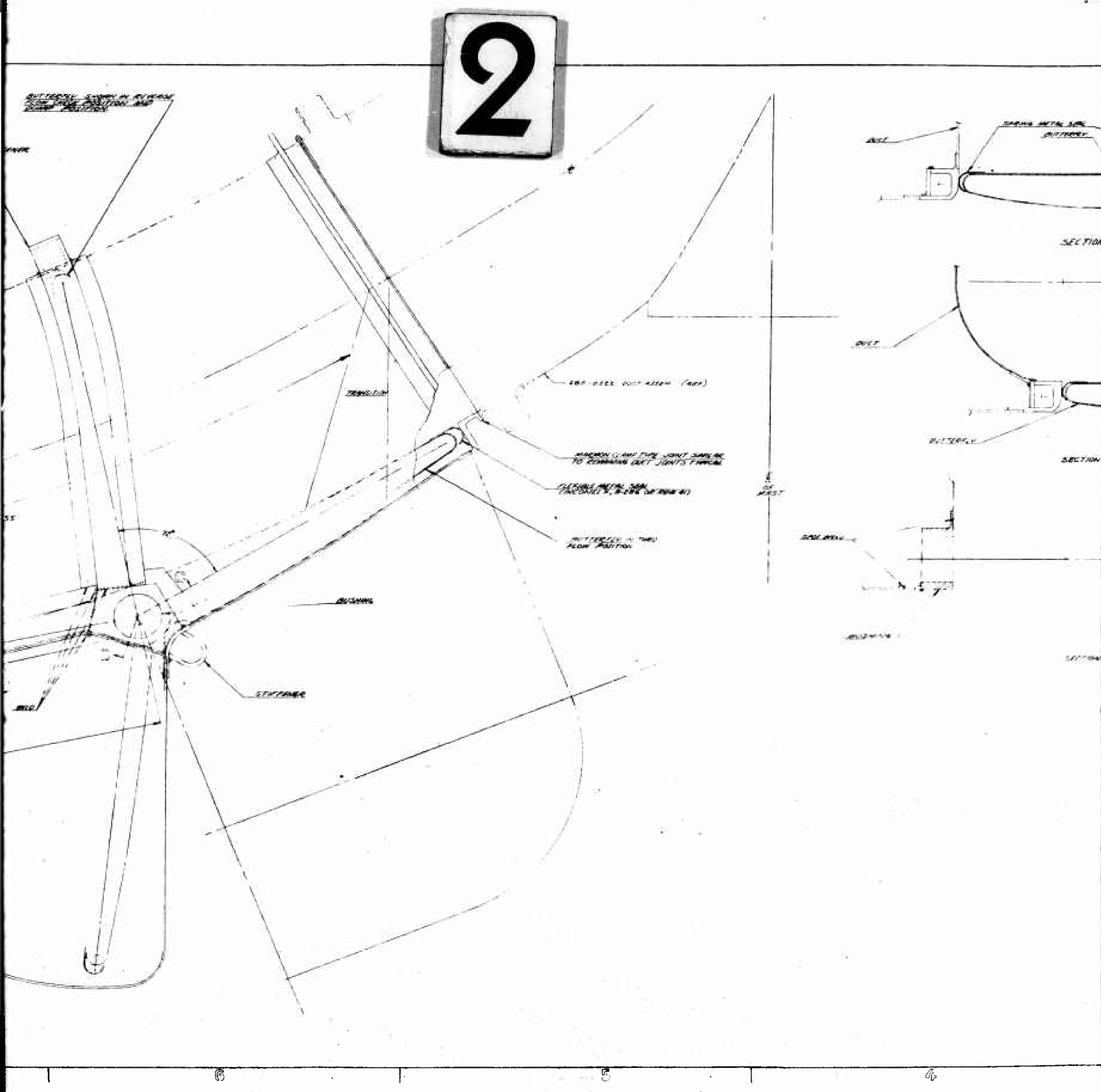
## 4.3 DESIGN AND EVALUATION OF DIVERTER VALVES

The preliminary design of five different diverter valves as conceived by this Contractor, plus a picture of the existing General Electric J85 diverter valve, are given here as Figures 4-2 through 4-7. Design studies of diverter valves were made by this Contractor for several reasons:

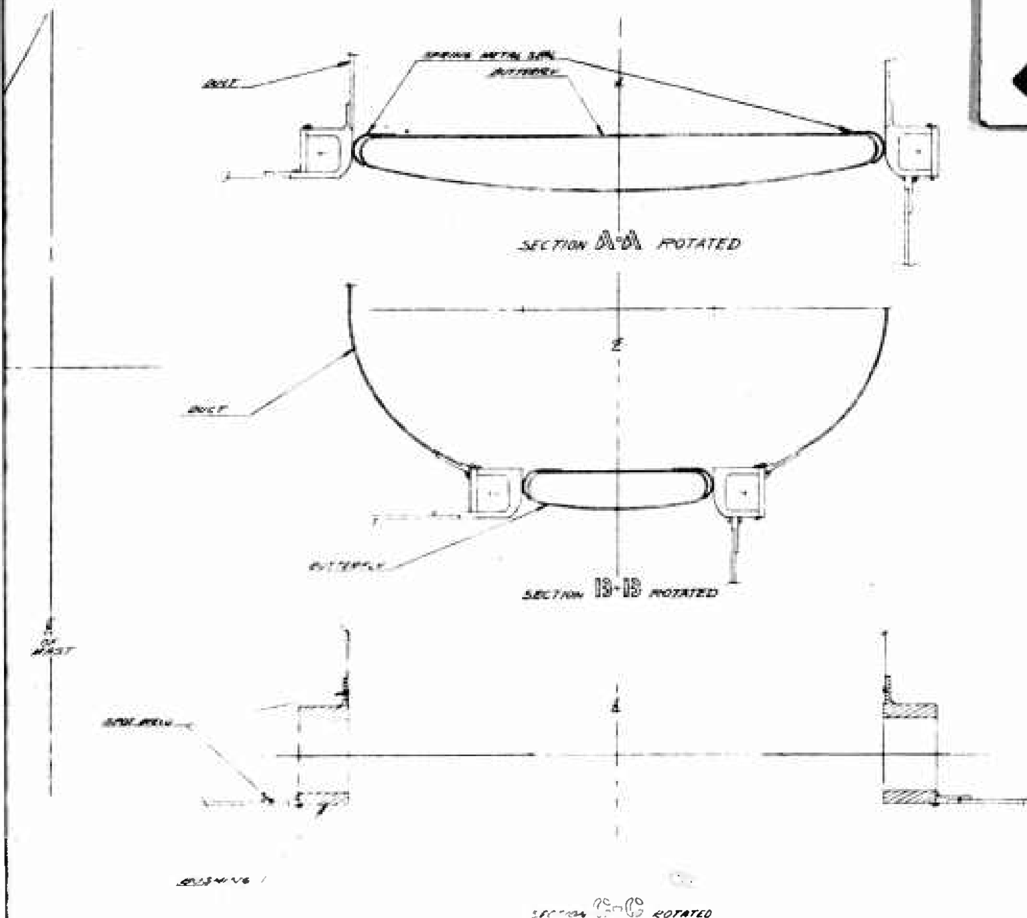
(1) No diverter valve exists specifically designed for the T64.

(2) Hot cycle helicopter propulsion configurations such as those in Figures 4-1a or 4-1b require minimum pressure drop only when gas is diverted to the rotor. Pressure drop is not important when gas is diverted overboard. Diverter valves which now exist, such as the General Electric J85 valve, or the Solar diverter valve for the Pratt & Whitney


1
BELLOWS
STIFFENER
DUCT
AREA OF FLATNESS
FLOW
STIFFENER
STIFFENER
8
7
6


2
DUCT
SPRING METAL SEAL
BUTTERFLY
SECTION
TRANSITION
BUSHING
STIFFENER
6
5
4

3



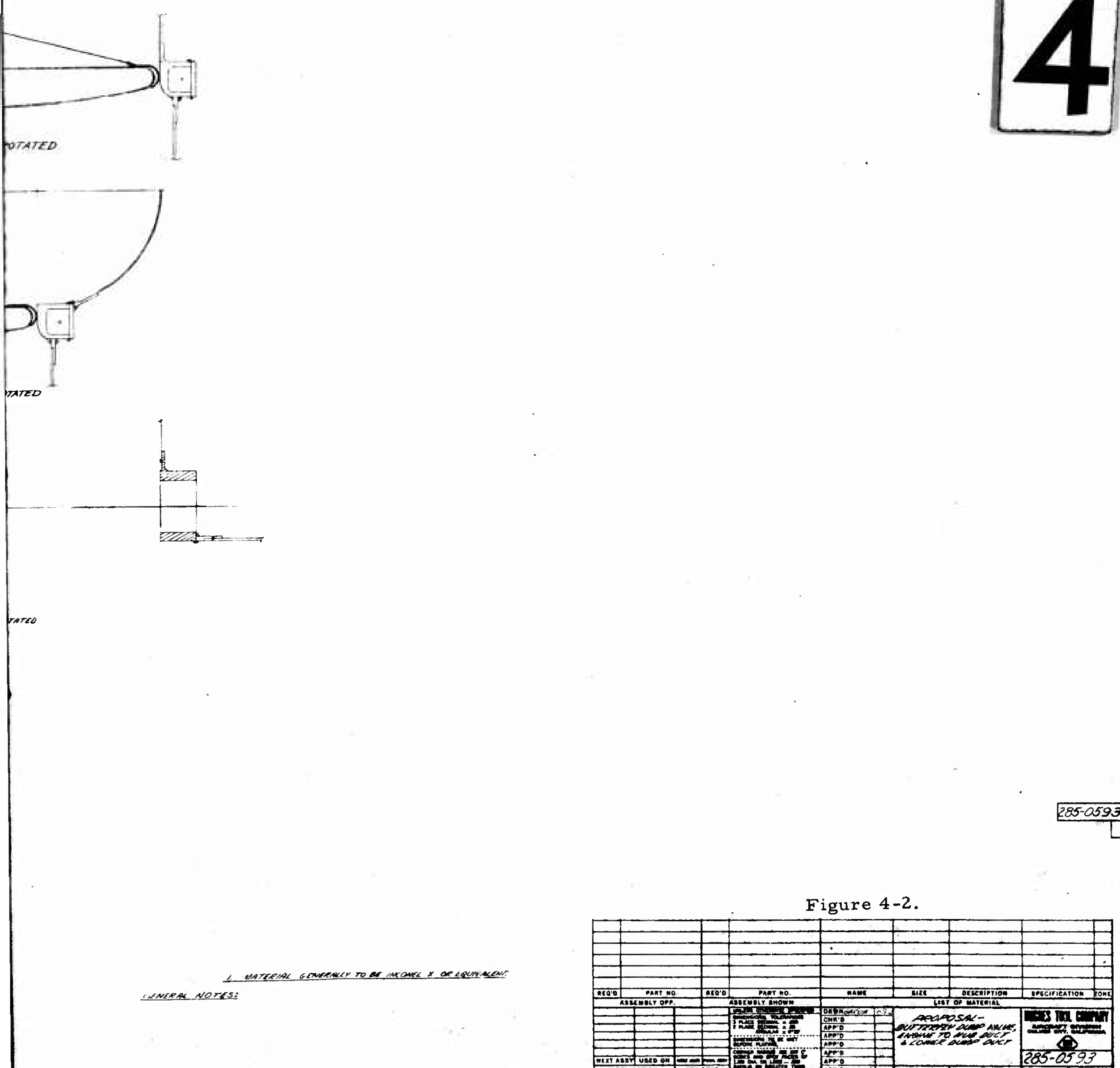

GENERAL NOTES:

1 MATERIAL GENERALLY TO BE INCONEL X OR EQUIVALENT

| REQ'D | PART NO. | REQ'D | PART NO. |
|---|---|---|---|
| ASSEMBLY OPP. | | ASSEMBLY SHOWN | |

| NEXT ASSY | USED ON | | |
|---|---|---|---|
| APPLICATION | | QTY REQD | |

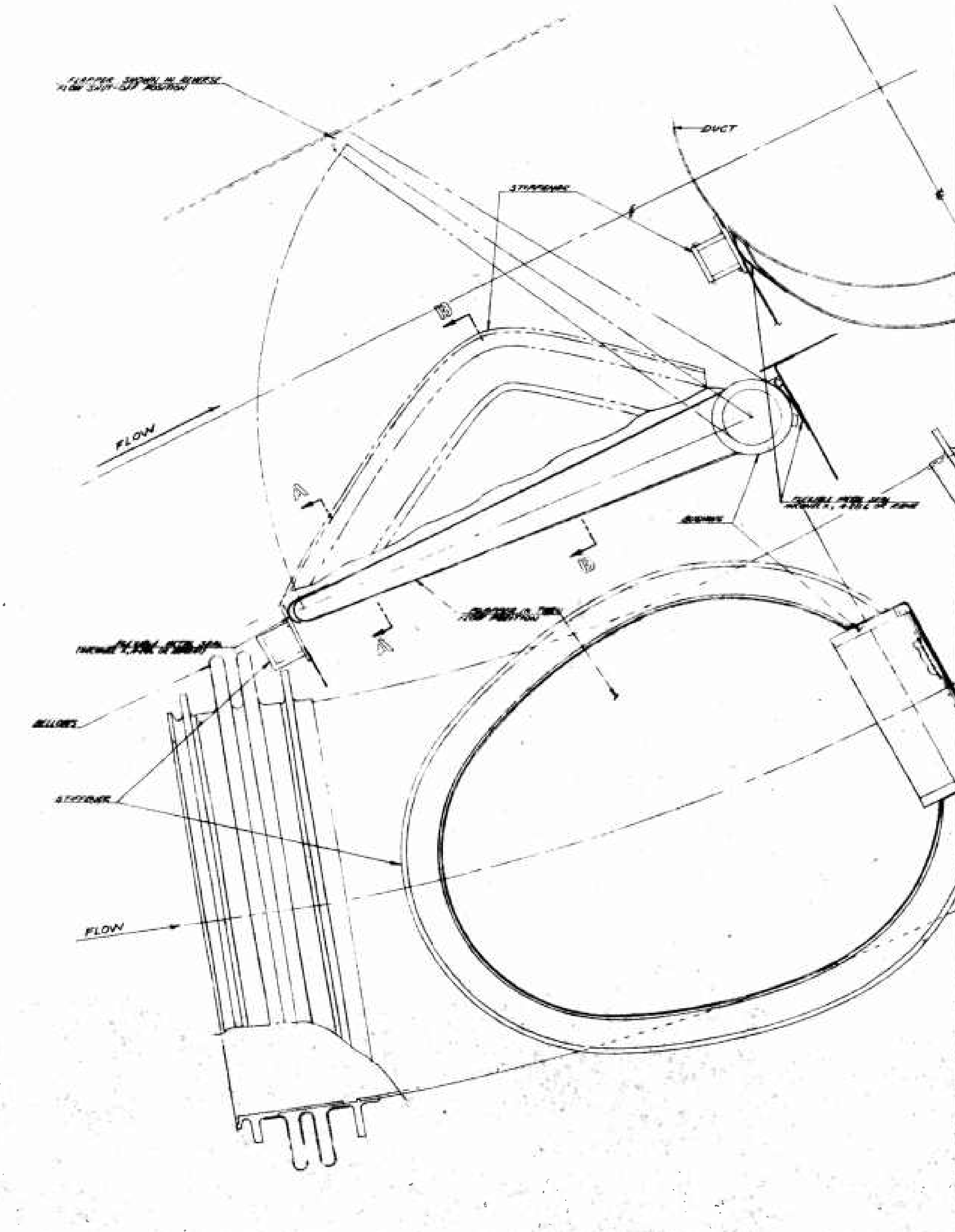
4



OTATED
TATED
TATED
1. MATERIAL GENERALLY TO BE INCONEL X OR EQUIVALENT
GENERAL NOTES:
285-0593
REQ'D
PART NO
REQ'D
PART NO.
NAME
SIZE
DESCRIPTION
SPECIFICATION
ZONE
ASSEMBLY OPP.
ASSEMBLY SHOWN
LIST OF MATERIAL
NEXT ASSY
USED ON
APPLICATION
QTY REQD
DRWN
CHK'D
APP'D
APP'D
APP'D
APP'D
APP'D
APP'D
PROPOSAL - BUTTERFLY DUMP VALVE, ENGINE TO HUB DUCT & LOWER DUMP DUCT
285-0593
SCALE 1/1
CODE 02731

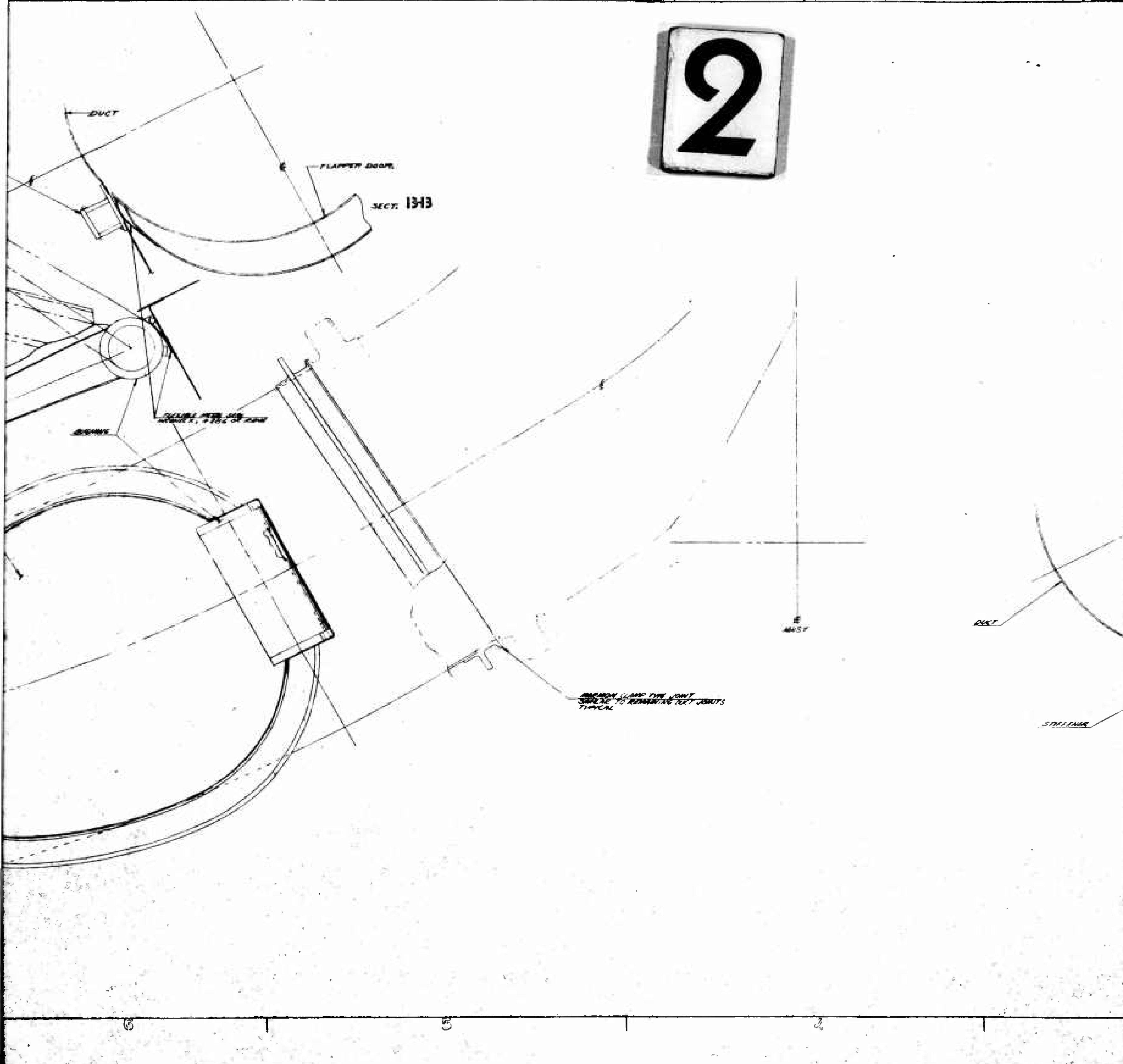

Figure 4-2.

1



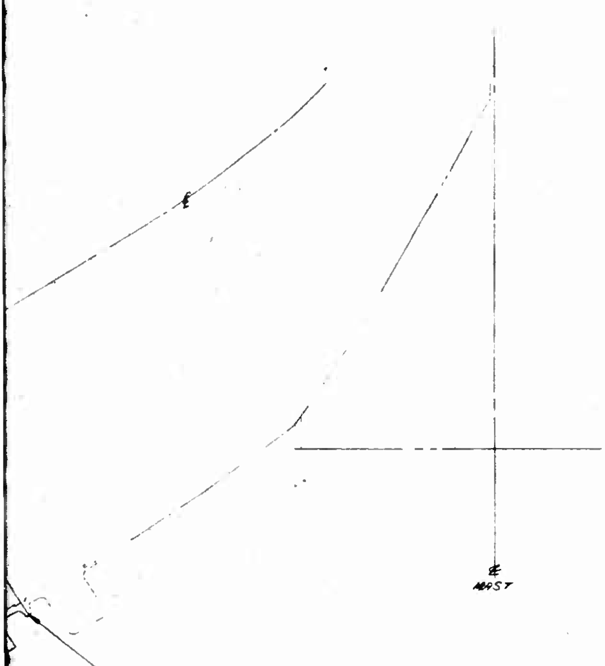

7

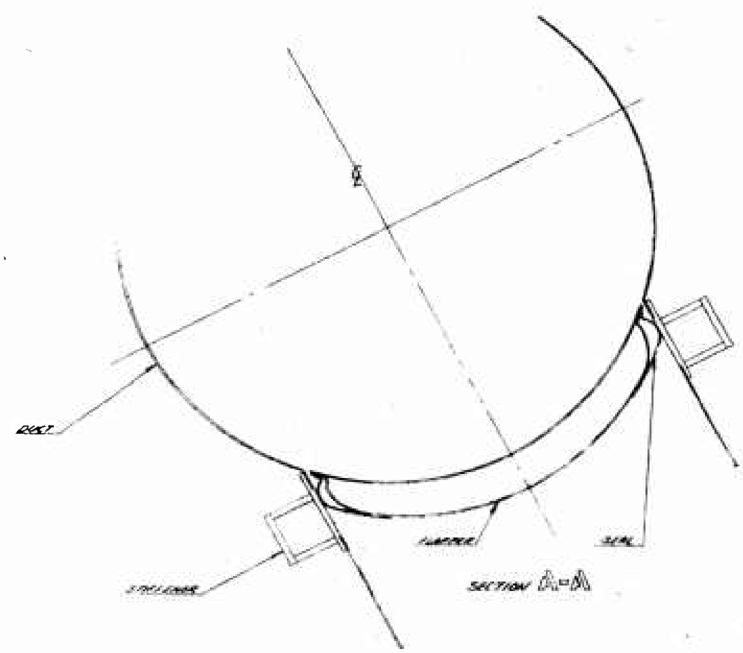
6



2
DUCT
FLAPPER DOOR
SECT. 13-13
CLAMP TYPE JOINT
TYPICAL
DUCT
STIFFENER
6
5
4

3




SECTION A-A

1 MATERIAL GENERALLY TO BE INCONEL X OR EQUIVALENT.

GENERAL NOTES:

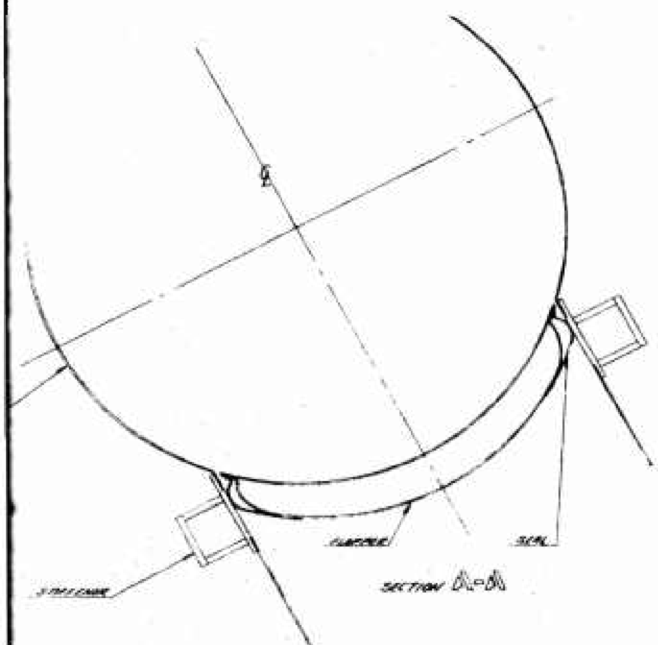
REQ'D

NEXT ASSY

APPLIC

4 3 2

4




285-0594

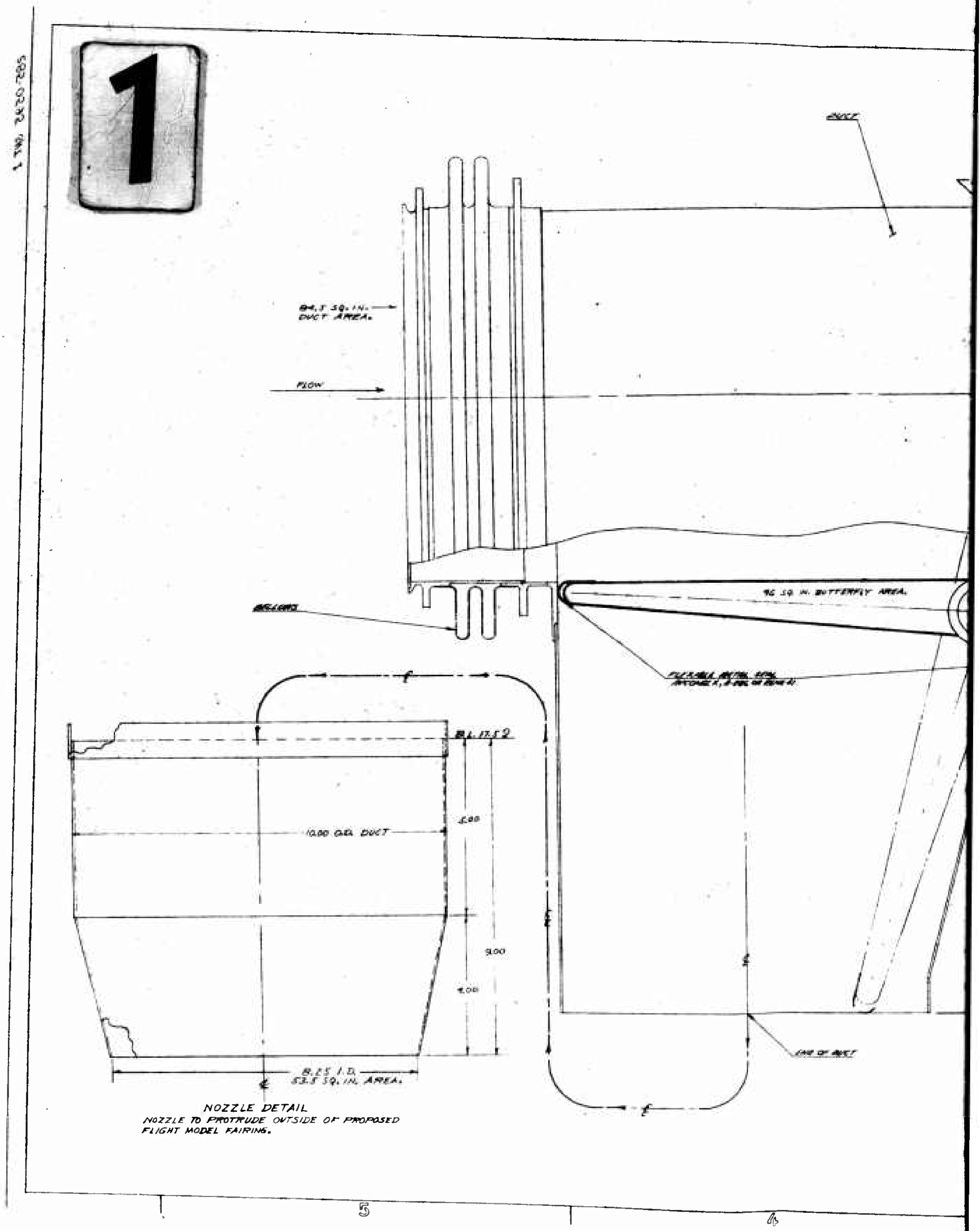
GENERAL NOTES:

1. MATERIAL GENERALLY TO BE INCONEL X OR EQUIVALENT.

Figure 4-3.

| REQ'D | PART NO. | REQ'D | PART NO. | NAME | SIZE | DESCRIPTION | SPECIFICATION | ZONE |
|---|---|---|---|---|---|---|---|---|
| ASSEMBLY OPP. | | ASSEMBLY SHOWN | | LIST OF MATERIAL | | | | |

| | | |
|---|---|---|
| NEXT ASSY / USED ON — APPLICATION; QTY REQD | DRWN / CHK'D / APP'D / APP'D / APP'D / APP'D / APP'D / APP'D | PROPOSAL — FLAPPER DUMP VALVE, ENGINE TO HUB DUCT & SIDE DUMP DUCT |
| SCALE FULL | | HUGHES TOOL COMPANY, AIRCRAFT DIVISION — 285-0594 — CODE 02731 |


1
DUCT
84.5 SQ. IN. DUCT AREA.
FLOW
BELLOWS
96 SQ. IN. BUTTERFLY AREA.
B.L. 17.52
10.00 O.D. DUCT
5.00
9.00
4.00
8.25 I.D.
53.5 SQ. IN. AREA.
NOZZLE DETAIL
NOZZLE TO PROTRUDE OUTSIDE OF PROPOSED FLIGHT MODEL FAIRING.
END OF DUCT
5

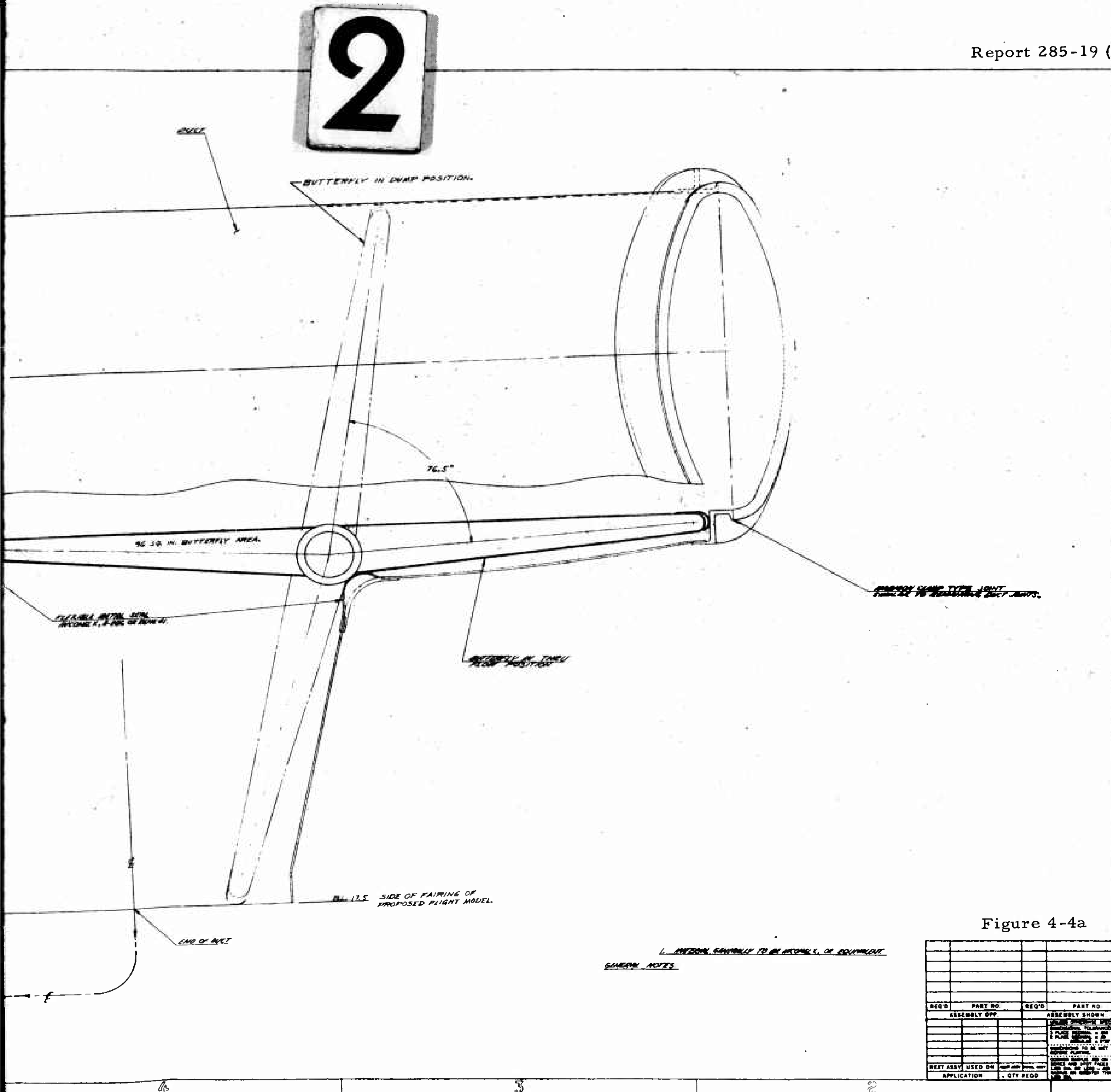

2
BUTTERFLY IN DUMP POSITION.
76.5°
96 SQ. IN. BUTTERFLY AREA.
BL 17.5 SIDE OF FAIRING OF PROPOSED FLIGHT MODEL.
END OF DUCT
GENERAL NOTES
REQ'D
PART NO.
ASSEMBLY OPP.
ASSEMBLY SHOWN
NEXT ASSY
USED ON
APPLICATION
QTY REQD

Figure 4-4a

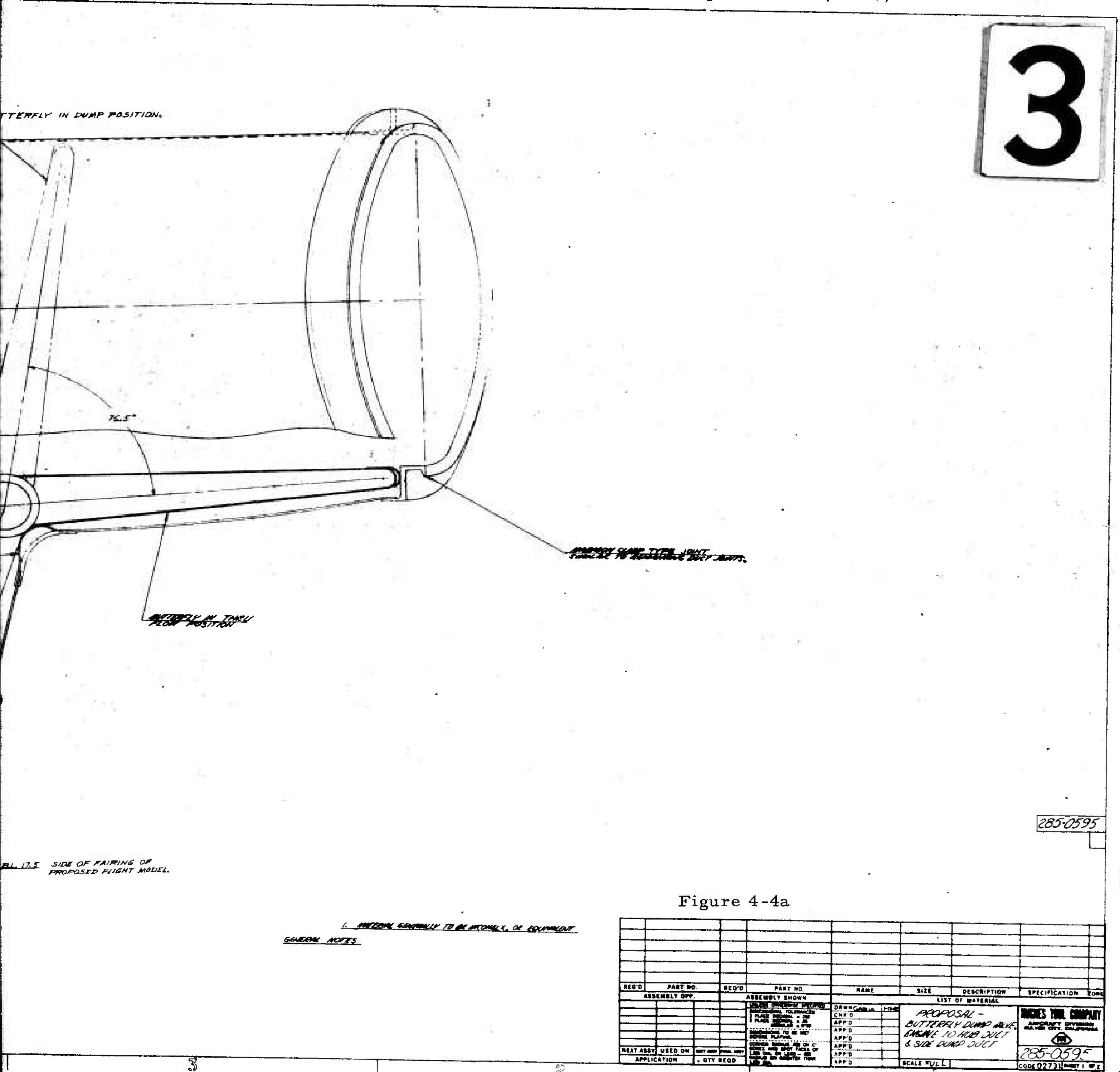
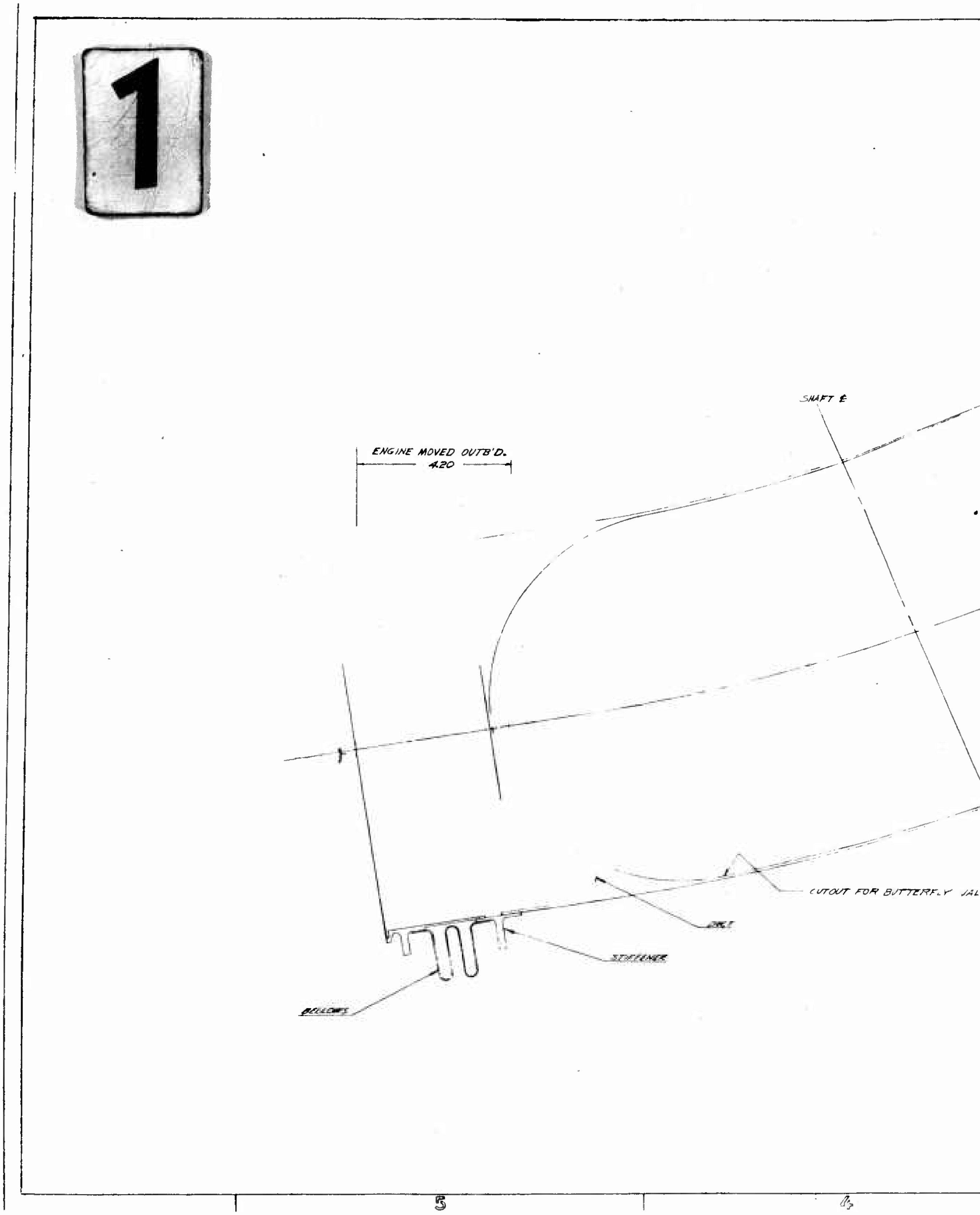
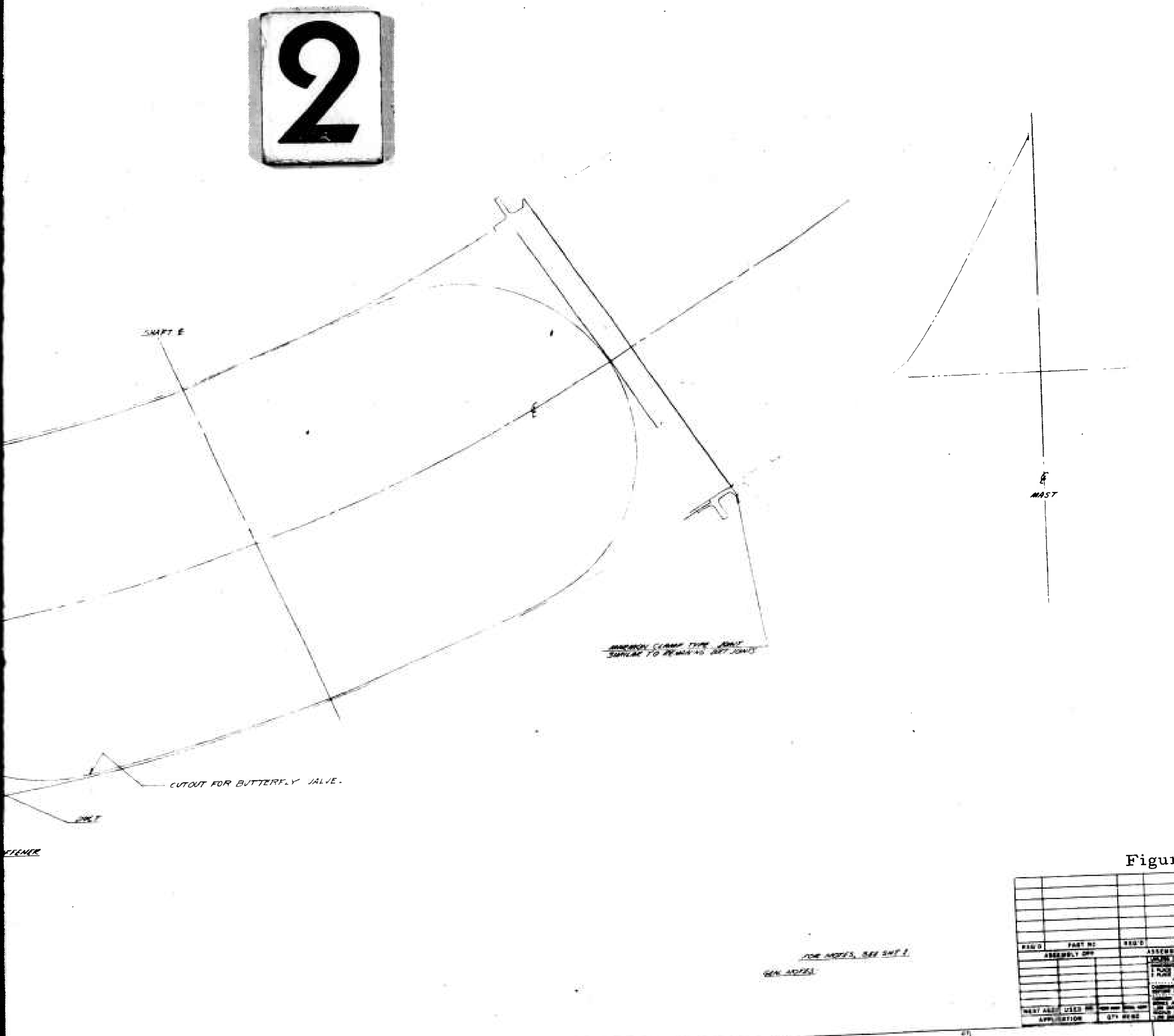

TTERFLY IN DUMP POSITION.
76.5°
BUTTERFLY IN THRU FLOW POSITION
BL 17.5 SIDE OF FAIRING OF PROPOSED FLIGHT MODEL.
285-0595
GENERAL NOTES
LIST OF MATERIAL
PROPOSAL - BUTTERFLY DUMP VALVE, ENGINE TO HUB DUCT & SIDE DUMP DUCT
HUGHES TOOL COMPANY
285-0595
SCALE FULL

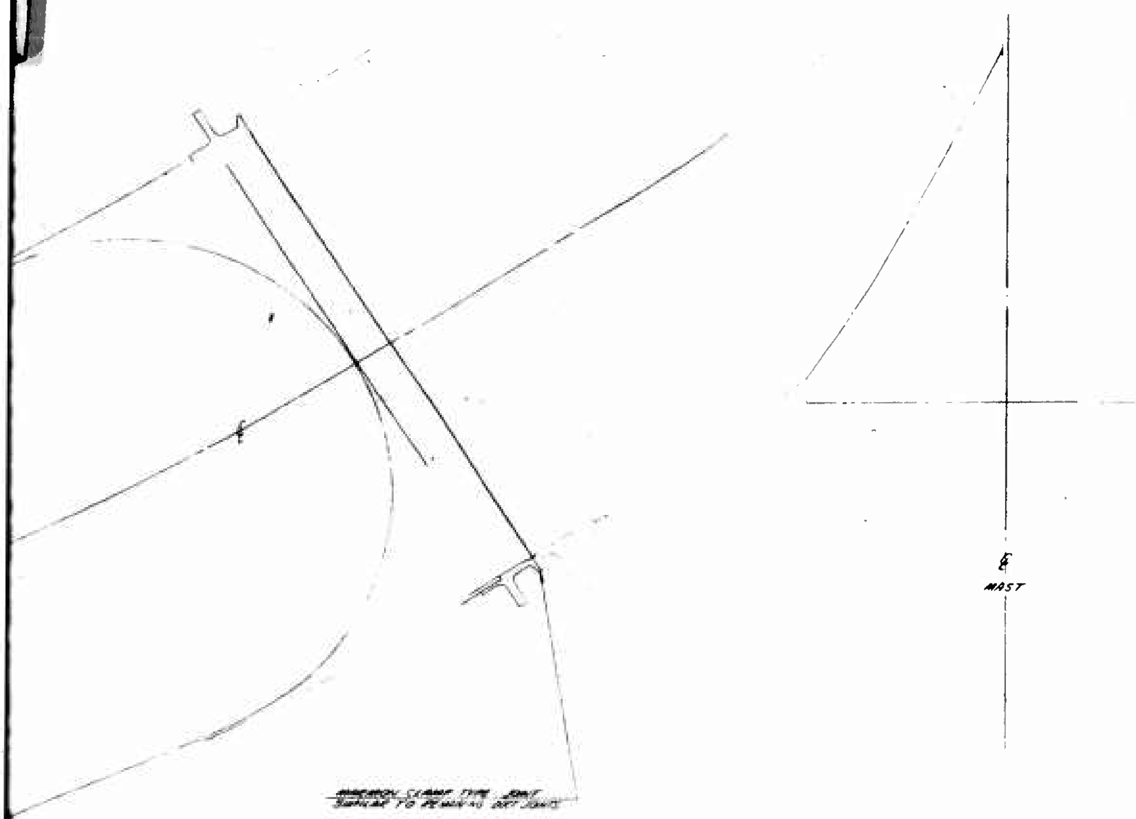
Figure 4-4a


SHAFT ℄
ENGINE MOVED OUTB'D.
4.20
CUTOUT FOR BUTTERFLY VAL
STIFFENER
BELLOWS


2
SHAFT ℄
℄
℄
MAST
CUTOUT FOR BUTTERFLY VALVE.
DUCT
FFENER
FOR NOTES, SEE SHT 1
GEN NOTES
REQ'D
PART NO
ASSEMBLY
APPLICATION

Figur

3




285-0595

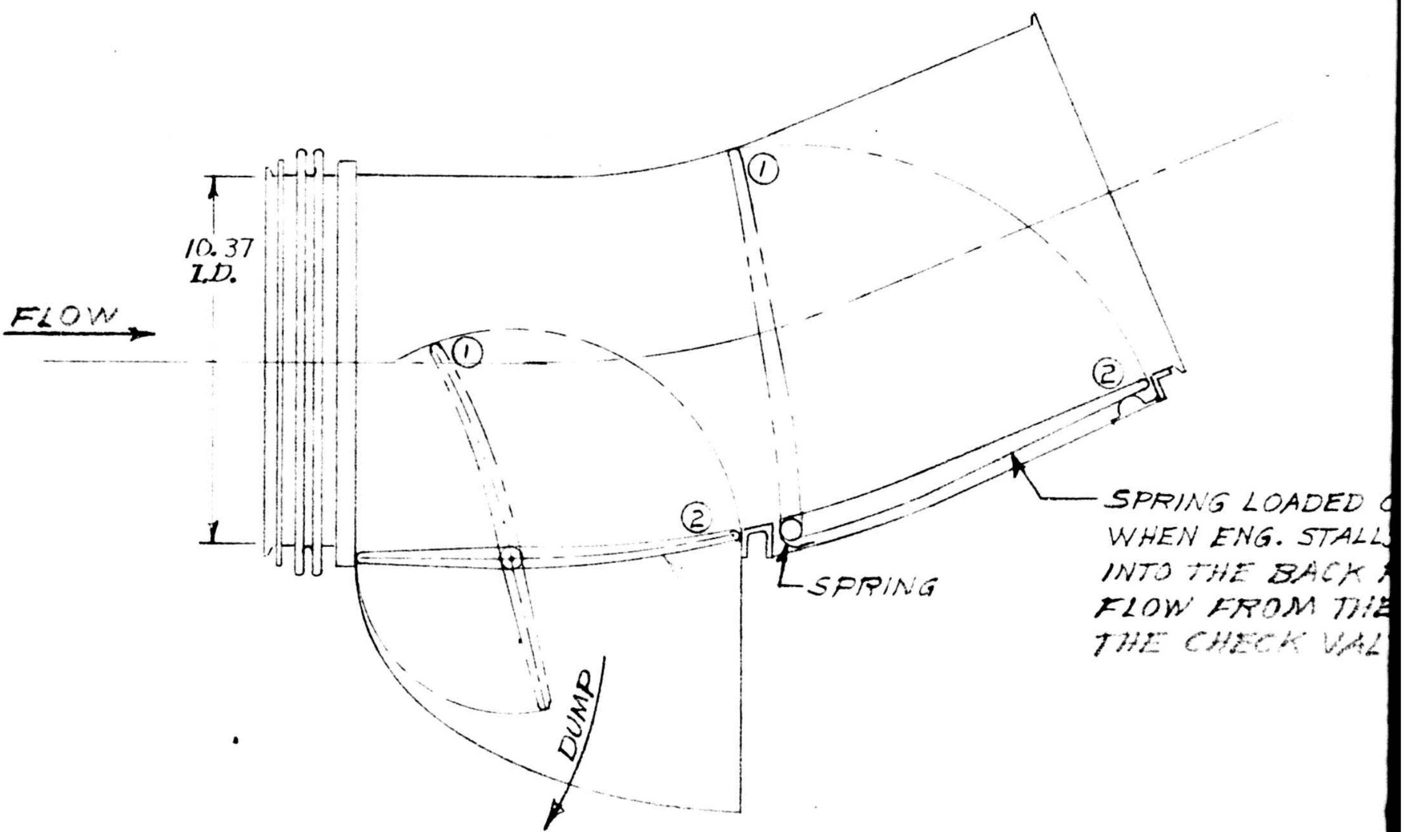
Figure 4-4b

| REQ'D | PART NO | REQ'D | PART NO | NAME | SIZE | DESCRIPTION | SPECIFICATION | ZONE |
|---|---|---|---|---|---|---|---|---|
| ASSEMBLY OPP | | ASSEMBLY SHOWN | | LIST OF MATERIAL | | | | |

| | | | |
|---|---|---|---|
| NEXT ASSY / USED ON / APPLICATION / QTY REQD | UNLESS OTHERWISE SPECIFIED. DIMENSIONAL TOLERANCES: 1 PLACE DECIMAL; 2 PLACE DECIMAL; ANGULAR. DIMENSIONS TO BE MET BEFORE PLATING. CORNER RADIUS ON C' BORES AND SPOT FACES OF 1.00 DIA. OR LESS; RADIUS ON GREATER THAN 1.00 DIA. | DRWN / CHK'D / APP'D / APP'D / APP'D / APP'D / APP'D / APP'D | PROPOSAL - BUTTERFLY DUMP VALVE, ENGINE TO HUB DUCT & SIDE DUMP DUCT |
| HUGHES TOOL COMPANY, AIRCRAFT DIVISION, CULVER CITY, CALIFORNIA | 285-0595 | SCALE | CODE 02731 SHEET 2 OF 2 |

1

DO NOT SCALE




| | | | | |
|---|---|---|---|---|
| | | | | |
| REQD | PART NO. | REQD | PART NO. | |
| ASSEMBLY OPP. | | ASSEMBLY SHOWN | | |
| | | | UNLESS OTHERWISE SPECIFIED | DR |
| | | | DIMENSIONAL TOLERANCES<br>3 PLACE DECIMAL ± .010<br>2 PLACE DECIMAL ± .03<br>ANGULAR ± 0°30' | CH<br>AP |
| | | | DIMENSIONS TO BE MET BEFORE PLATING. | AP<br>AP |
| NEXT ASSY | USED ON | NEXT ASSY | FINAL ASSY | CORNER RADIUS .062 ON C' BORES AND SPOT FACES OF 1.250 DIA. OR LESS — .093 RADIUS ON GREATER THAN 1.250 DIA. |
| APPLICATION | | QTY REQD | | AP<br>AP<br>AP |

DIETERICH-POST CLEARPRINT 1000H-10

285-0597

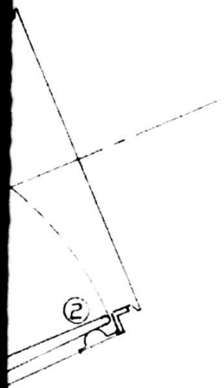
**REVISIONS**

| SYM | E.O.'S | DESCRIPTION | DRWN | APP'D | DATE |
|---|---|---|---|---|---|
| | | | | | |

ACTUATING SEQUENCE.

A.) BEFORE STARTING, BOTH DOORS ARE MOVED BY THE ACTUATOR TO POSITION ①.

B.) FOR POWERING ROTOR, BOTH DOORS ARE MOVED TO POSITION ②.




2

SPRING LOADED CHECK VALVE.
WHEN ENG. STALLS, A SPRING KICKS THE VALVE INTO THE BACK FLOW FROM THE OPPOSITE ENG. FLOW FROM THE OPPOSITE ENG. THEN FORCES THE CHECK VALVE TO A CLOSED POSITION.

Figure 4-6
Report 285-19 (62-19)

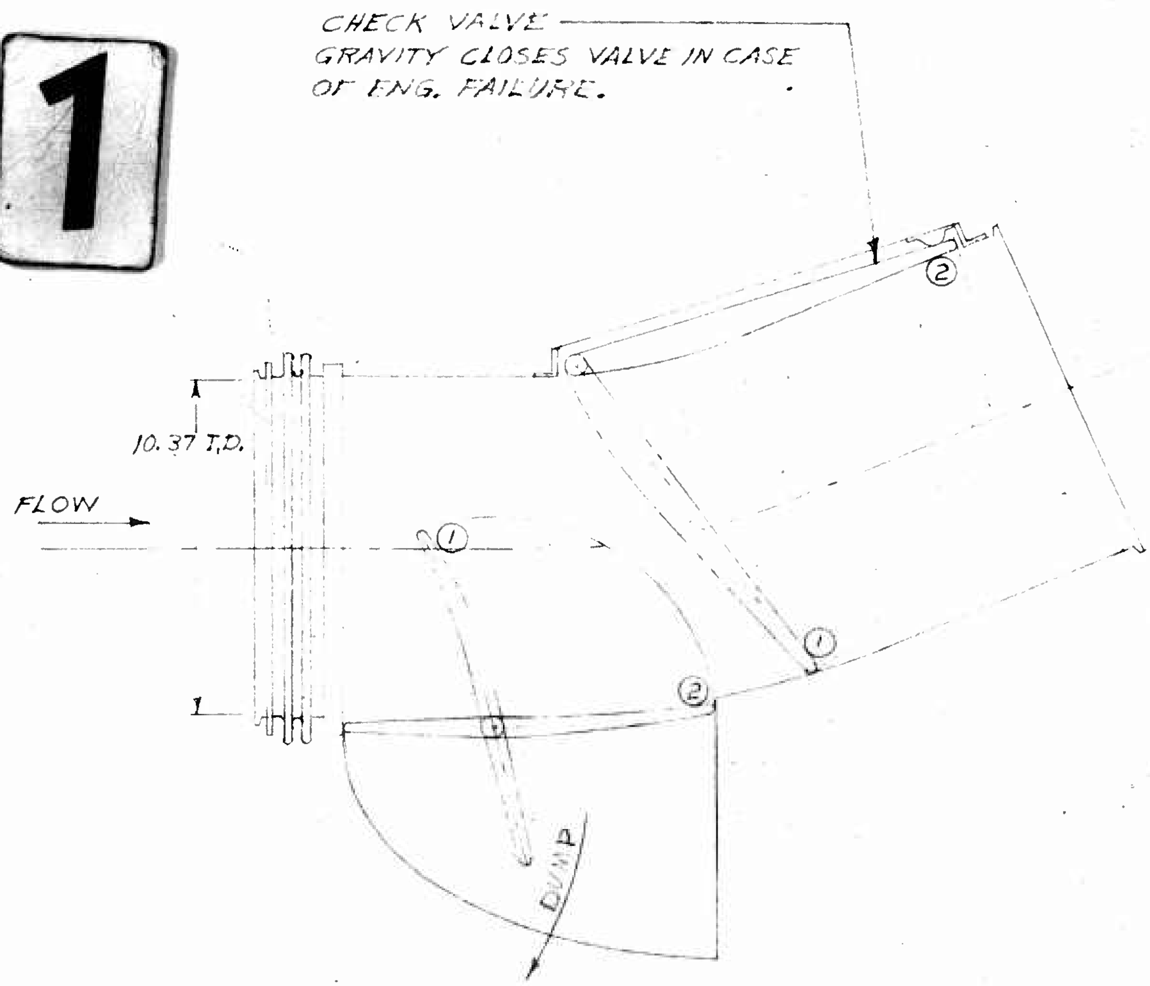
| | | | | |
|---|---|---|---|---|
| | | | | |
| PART NO. | NAME | SIZE | DESCRIPTION | SPECIFICATION |
| LY SHOWN | LIST OF MATERIAL | | | |

| | | | | |
|---|---|---|---|---|
| THERWISE SPECIFIED | DRWN Sallows | 3-14-62 | PROPOSAL - DUMP & LOWER CHECK VALVE, ENG. TO HUB DUCT & SIDE DUMP DUCT. | HUGHES TOOL COMPANY<br>AIRCRAFT DIVISION<br>CULVER CITY, CALIFORNIA |
| NAL TOLERANCES<br>DECIMAL ± .010<br>DECIMAL ± .03<br>NGULAR ± 0°30' | CHK'D | | | |
| | APP'D | | | |
| NS TO BE MET<br>LATING. | APP'D | | | |
| | APP'D | | | |
| RADIUS .062 ON C'<br>D SPOT FACES OF<br>OR LESS — .093<br>N GREATER THAN | APP'D | | | 285-0597 |
| | APP'D | | | |
| | APP'D | | SCALE 1/50 | CODE 02731 SHEET OF |

VELLUM "B" SIZE FORM 9487 REV. 1-61

1

DO NOT SCALE




| | | | |
|---|---|---|---|
| | | | |
| REQD | PART NO. | REQD | PART NO. |
| ASSEMBLY OPP. | | ASSEMBLY SHOWN | |

| | | | |
|---|---|---|---|
| | | | |
| | | | |
| | | | |
| | | | |
| | | | |
| | | | |
| NEXT ASSY | USED ON | NEXT ASSY | FINAL ASSY |
| APPLICATION | | QTY REQD | |

UNLESS OTHERWISE SPECIFIED

DIMENSIONAL TOLERANCES
3 PLACE DECIMAL ± .010
2 PLACE DECIMAL ± .03
ANGULAR ± 0°30′

DIMENSIONS TO BE MET BEFORE PLATING.

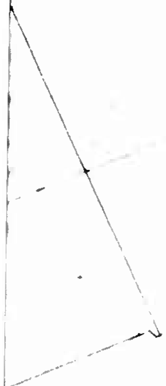
CORNER RADIUS .062 ON C' BORES AND SPOT FACES OF 1.250 DIA. OR LESS — .093 RADIUS ON GREATER THAN 1.250 DIA.

DRW
CHK
APP
APP
APP
APP
APP
APP

285-0596

| REVISIONS | | | | | |
|---|---|---|---|---|---|
| SYM | E.O.'S | DESCRIPTION | DRWN | APP'D | DATE |
| | | | | | |

**2**



ACTUATING SEQUENCE.

A.) BEFORE STARTING, BOTH DOORS ARE MOVED BY THE ACTUATOR TO POSITION (1).

B.) FOR POWERING ROTOR, BOTH DOORS ARE MOVED TO POSITION (2).

Figure 4-5.
Report 285-19 (62-19)

| PART NO. | NAME | SIZE | DESCRIPTION | SPECIFICATION |
|---|---|---|---|---|
| LY SHOWN | LIST OF MATERIAL | | | |

| | | | | |
|---|---|---|---|---|
| OTHERWISE SPECIFIED | DRWN Sallows | 2-14-62 | PROPOSAL - DUMP & OVERHEAD CHECK VALVE, ENG. TO HUB DUCT & SIDE DUMP DUCT. | HUGHES TOOL COMPANY AIRCRAFT DIVISION CULVER CITY, CALIFORNIA |
| NAL TOLERANCES DECIMAL ± .010 DECIMAL ± .03 ANGULAR ± 0°30' | CHK'D | | | |
| | APP'D | | | |
| NS TO BE MET PLATING. | APP'D | | | |
| | APP'D | | | 285-0596 |
| RADIUS .062 ON C' ND SPOT FACES OF OR LESS — .093 N GREATER THAN | APP'D | | | |
| | APP'D | | | |
| | APP'D | | SCALE 1/50 | CODE 02731 SHEET OF |

VELLUM "B" SIZE FORM 9487 REV. 1-61

ANALYSIS
PREPARED BY
CHECKED BY

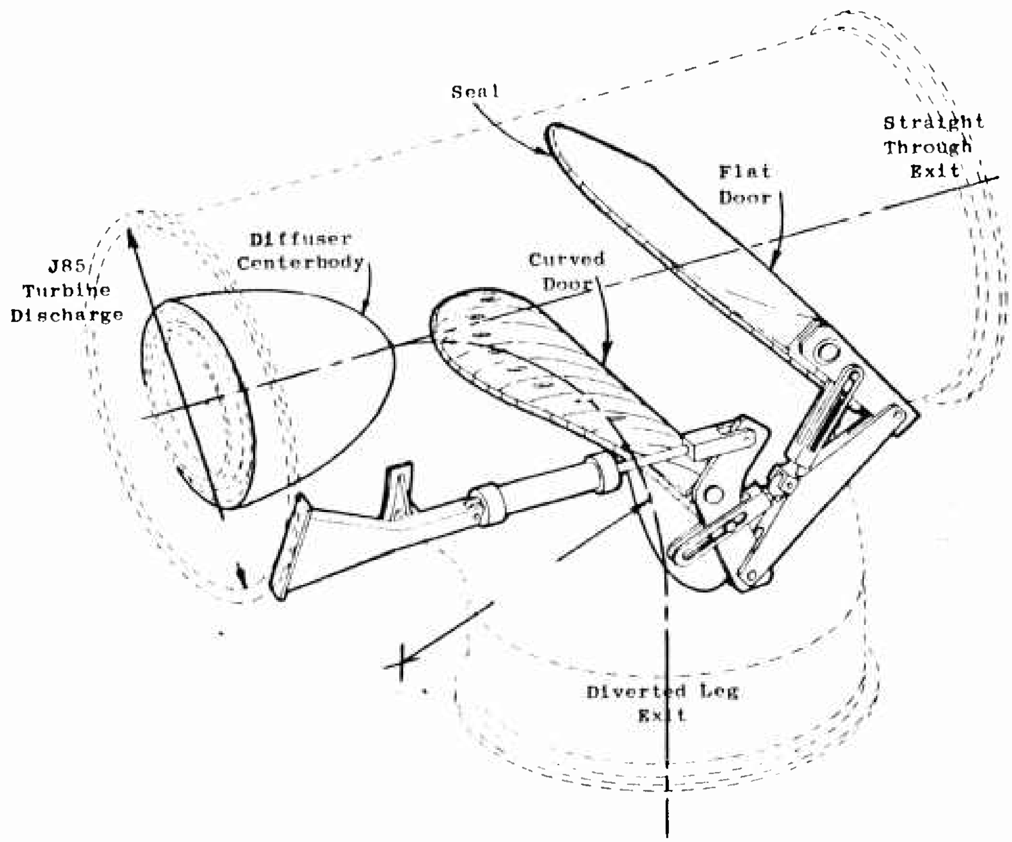



Figure 4-7. General Electric J85 Diverter Valve

J60 in the Lockheed Hummingbird, usually have very low pressure drop in either position of the valve. As a result, an extra aerodynamic factor enters the picture, making valves such as those from General Electric or Solar more refined than they really need to be for many hot cycle helicopter applications. In addition, these particular valves are designed for engines which handle almost twice as much air as the T64. As a result, the valves are really bigger and heavier than they need to be to handle T64 gas at reasonable pressure drops. A properly sized T64 valve will have a cross-section diameter only about 70% of the J85 or J60 valves.*

For these reasons, the Contractor made the preliminary designs shown in Figures 4-2 through 4-6, any of which can be used for the propulsion arrangements of Figures 4-1a or 4-1b. Comments on these designs follow:

### 4.3.1 Figure 4-2 (HTC Drawing 285-0593)

This drawing shows an elevation of a diverter valve with a single door hinged in the middle on a lateral shaft on the bottom of the gas duct. The door is slightly curved in the upward direction, following the contour of the duct. The door is sealed around its periphery with metal springs. The gas path in the straight through portion to the rotor is very clean and when the door is moved by an actuator (not shown here, but part of the over-all system), the gas is forced downward out of the main duct into a lower dump duct. The gas is then exhausted laterally through a nozzle at the right or to the left as suits the installation. Naturally this devious path for the overboard dump has a relatively high pressure drop, but it is unimportant, as mentioned in Section 4.2. The only effect of high pressure drop is to require a somewhat larger nozzle area on the dump leg. This dump exhaust area may be geometrically 15-20% higher than that at the blade tips, but the effective areas in the two directions (rotor or overboard) should be nearly equal to keep the engine on the same operating line.

The door for this design will also serve as a check valve when moved to the "gas overboard" position. Flow from the other engine will not be able to go backwards through a dead or shut off engine when the diverter valve door is moved to close off the main duct.

* The actual YT64 to be used in the forthcoming whirl test will have an oversize tail diameter for expediency. As a result, an oversize diverter valve may actually be preferable.

4.3.2 Figure 4-3 (HTC Drawing 285-0594)

This drawings shows the elevation of a diverter valve with a single door hinged at one edge on a shaft at the side of the gas duct. This valve is very clean in the straight-through direction. Gas is directed overboard when the door is moved across the duct. Unlike the design of Figure 4-2, the valve shown here would direct gas overboard directly to the side of the duct without going first through a lower dump duct. Seals similar to Figure 4-2 are used. The valve would also act as a check valve in the diverted position. Because this door is hinged at the edge rather than in the middle, as the Figure 4-2 designs, actuation loads against gas pressure will be substantially higher.

4.3.3 Figure 4-4a and 4-4b (HTC Drawing 285-0595, Sheets 1 and 2)

These drawings are the plan and elevation of a diverter valve with a single door hinged in the middle on a shaft at the side of the duct. This design is similar to Figure 4-2, but it has a side dump duct rather than a lower dump duct. Also, the door is not straight on each side of the shaft. This difference may lead to a little harder sealing problem because the duct-door clearance in transition is a variable, rather than a constant, as was the Figure 4-2 design.

4.3.4 Figure 4-5 (HTC Drawing 285-0596)

This drawing shows the elevation of a valve with two doors instead of one. The use of two doors adds a great deal of flexibility to the utility of the diverter valve. In Section 3.2.7, it was pointed out that power recovery from autorotation would take about six seconds. If the total nozzle effective area is increased, the gas generator idle speed will increase cutting down the time to recover power on demand, compared to the present engine configuration. The two door approach shown here could easily be used to increase greatly the nozzle effective area by opening both gas paths at the same time, thus reducing the engine back pressure, and speeding up the gas generators. The single door designs of Figures 4-2, 4-3, and 4-4 can accomplish this same increase of effective area by making the geometric nozzle area in the overboard leg much larger than the nozzle area in the straight through case. However, the valve would have to be fully diverted to take advantage of this increase, and a certain time lag would be required to come back to normal area for powered flight. Such a lag would be reduced with this two door approach.

This design also will "fail safe" if a door link breaks or a hydraulic failure occurs if a single hydraulic system is used. By "fail safe"

is meant automatic motion of the diverter valve doors to the straight through or rotor position if such failures occur.

No door latches are required with the two door approach because normal gas flow will move the doors out of the way.

When two doors are used and the overboard exhaust is to the side, the length of door that extends outward from the valve will be less than for the Figure 4-4 valve. Consequently, the fairing over the engines can be narrower, reducing weight and drag somewhat.

### 4.3.5 Figure 4-6 (HTC Drawing 285-0597)

This drawing shows a valve design which essentially duplicates Figure 4-5, but has the check valve hinged at the top of the duct instead of the bottom. Thus, gravity will help move the check valve in case of a link or hydraulic system failure.

### 4.3.6 Figure 4-7 (General Electric J85 Valve)

This figure is a cut away drawing of the General Electric diverter valve designed for the J85 turbojet engine. The valve has been built and tested and will be given flight qualification tests as part of the propulsion system for the General Electric VTOL airplane, the VZ11 (being built by Ryan under subcontract to G. E.). This valve is designed to produce minimum pressure drop in either position. It has two doors with interconnecting linkage. A complete description of the valve and results of tests are given in Reference 13.

The G. E. J85 valve can be used for engine-diverter valve arrangements of Figures 4-1a and 4-1b, just as any of the proposed diverter valves of Figures 4-2 through 4-6. Because of its size, which is still somewhat smaller (16.5 inches diameter) than the back of the YT64 to be used in the forthcoming whirl test, (18.0 inches diameter), it will have a lower flow Mach number than would a diverter valve of optimum size (11 inch diameter) for the T64. Consequently, the pressure drop for flow in the straight-through position will be quite low and very little higher even for the diverted position.

This low pressure drop in the diverted position permits use of the J85 diverter valve in the engine-diverter valve arrangement shown in Figure 4-1c. In this arrangement, the gases are directed to the rotor when the valve is in the diverted position. All of the other designs shown here would have had too high a pressure drop to permit this arrangement, which leads to certain other installation advantages.

In the first place, the distance from the centerline of the rotor to the furthest forward point over the top of the engine is several feet shorter than obtainable with any of the simpler Hughes designs. Consequently, a potential problem of a blade striking an engine is reduced. The engine center of gravity is moved closer to the rotor centerline, making it easier to balance helicopters that have forward-located engines.

In addition, the J85 valve, used in the Figure 4-1c arrangement has one feature which is quite different from all other valve designs examined so far The straight through position of the valve which is used to direct gas overboard can also be used as a propulsive jet. The T64 will produce a static thrust of almost 1800-1900 pounds, and later versions of the engine now being developed will produce perhaps as much as 2200 pounds static thrust. Two engines will produce 4400 pounds static thrust. This thrust can be used to propel a helicopter just like an autogyro. If the helicopter fuselage is sufficiently clean aerodynamically, and with the rotor in autorotation, the maximum lift-drag ratio of the helicopter will be high enough at high speed to permit rather high speed autogyro flight. Speeds of 150 knots or higher may be possible.

Since this speed range is above the normal helicopter speed range, very useful rotor load information can be obtained which can be used in designing rotors for compound helicopters that will eventually operate in the 200-275 knot speed range. Since this high speed autogyro feature will be obtainable from a helicopter at no expense except judicious arrangement of the engine-diverter valves, it represents a very strong reason for seriously considering use of J85 diverter valves in the Figure 4-1c propulsion arrangement. Further work will be necessary to confirm the benefits suggested by the preliminary work done under this contract.

A summary of the features of each of the six diverter valve designs considered in this study is given below in Table 4-1.

The several factors listed in Table 4-1 will aid in selecting a diverter valve design for a particular application. It should be noted that preliminary performance and weight estimates were made to insure that any of these valves would be satisfactory if they suit the application. If the propulsion arrangement is similar to Figures 4-1a or 4-1b and funds are available to develop an optimum diverter valve, it is expected that one of the valves of Figures 4-2 through 4-6 be chosen. If limited funds are available, the G.E. valve of Figure 4-7 would be acceptable. If the propulsion arrangement is to be the autogyro arrangement of Figure 4-1c, then the G.E. valve of Figure 4-7 would be chosen because this valve uniquely favors the autogyro propulsion arrangement.

## TABLE 4-1

## COMPARISON OF DIVERTER VALVE DESIGNS

| Figure No. | 4-2 | 4-3 | 4-4 | 4-5 | 4-6 | 4-7 |
|---|---|---|---|---|---|---|
| No. of Doors | 1 | 1 | 1 | 2 | 2 | 2 |
| Approximate Inlet Dia. | 10.9" | 10.9" | 10.9" | 10.9" | 10.9" | 16.5" |
| Location of Overboard Exhaust | Down and then Sideways | Sideways | Sideways | Down and then Sideways or Sideways | Down and then Sideways or Sideways | Sideways or Straight Through |
| Weight | Medium | Lowest | Medium | Medium | Medium | Highest |
| Pressure Drop In Rotor Direction | Min. | Min. | Min. | Min. | Min. | 1.25 x Min. |
| Actuation Force | Min. | Highest | Min. | Min. | Min. | Min. |
| Leakage | Min. | Min. | Min. | Min. | Min. | Highest |
| Linkage Complexity | Lowest | Lowest | Lowest | Highest | Highest | Medium |
| Status | Design | Design | Design | Design | Design | Available On Order |

The Phase I study which was concerned with the dynamics of the rotor-fuel control system was based on the assumption that diverter valves directed gas to the rotor with zero leakage and an average pressure drop of about 1.5% of engine total pressure. The several valve designs studied here will have minor variations in pressure drop and leakage from the assumed values. It is not expected that these minor variations will markedly affect the results of that Phase I dynamic study.

## 4.4 DESIGN AND EVALUATION OF BLADE DUCT VALVES

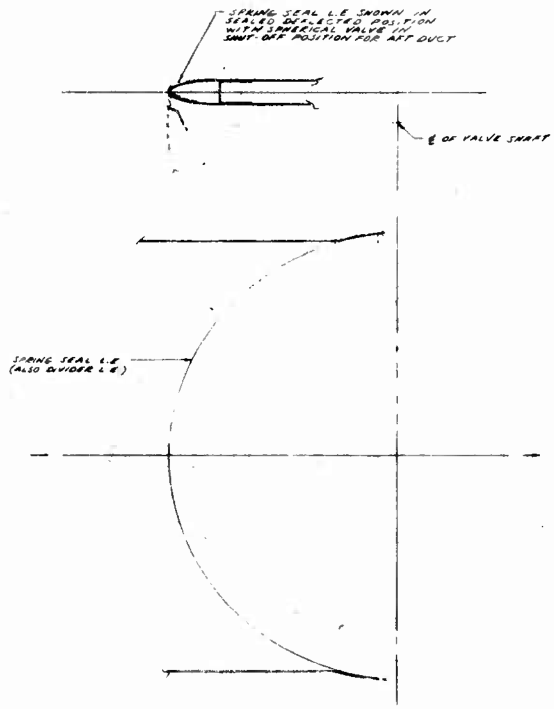
Three representative blade duct valves are shown in Figures 4-8, 4-9 and 4-10 which were designed to fulfill the requirements outlined under Paragraph 4.1. Comments on these three valve designs follow.

### 4.4.1 Figure 4-8 (HTC Drawing 285-0229)

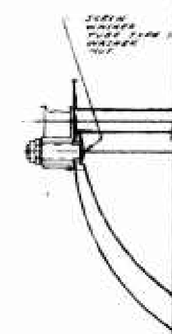
This drawing shows the plan view of a blade root duct valve which is designed to shut off one blade duct at the root following failure of one engine. This ball-type valve will have an actuation torque determined by friction. Gas pressure loads pass through the hinge of the valve and will not affect the actuation torque. This valve does have the disadvantage in its shutoff position of causing a pressure drop penalty because of the relationship of the flow divider and the duct valve to the flow from the remaining good engine. The geometry of the valve and flow splitter intersection leave the flow splitter protruding forward, requiring the flow streamlines to crowd abruptly together and undoubtedly causing some separation and pressure drop. It is possible that a flat door hinged upstream at the side of the duct could be used to shut off one duct, at the same time sealing at the leading edge of the flow divider instead of slightly downstream as the valve of Figure 4-8. However, this proposed door type valve would have very high actuation forces. Preliminary work indicates a very difficult seal problem is encountered, but perhaps further work will reveal a practical solution.

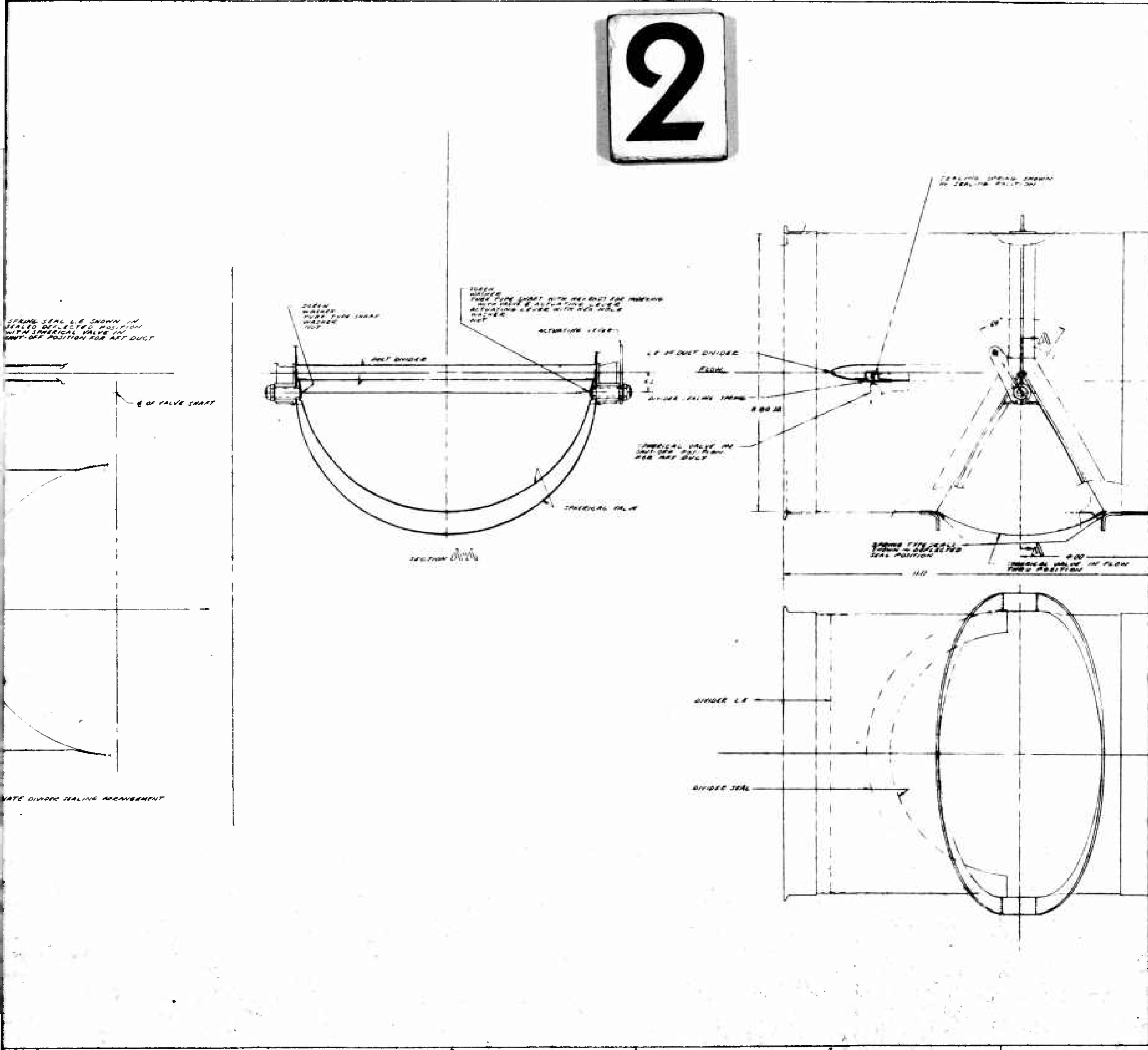
### 4.4.2 Figure 4-9 (HTC Drawing 285-0230)

This drawing shows the plan view of a blade root duct shutoff valve that is located on the center line of the duct, with a downstream hinge and with its upstream edge located behind a fixed flow splitter. The splitter has a detent device in it which serves to hold the flow splitter steady. This type of valve is an unstable device; i.e., once it leaves its neutral position it wants to continue to close instead of returning to neutral. Further, the door might tend to flutter if the leading edge were not shadowed by the flow splitter. The actuation forces on this type of door will be small because of

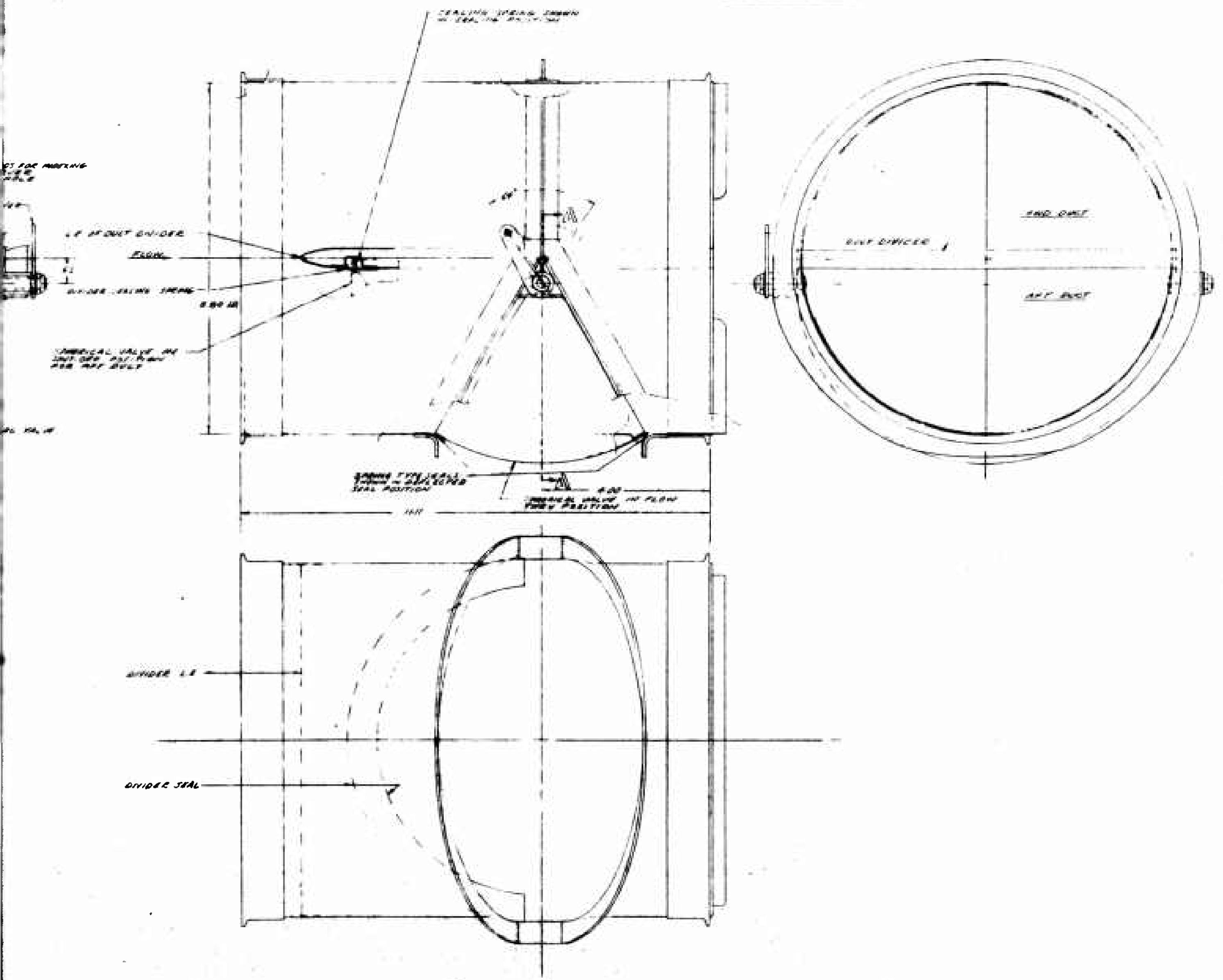



DETAILS OF ALTERNATE DIVIDER SEALING ARRANGEMENT




2
SPRING SEAL L.E. SHOWN IN SEALED DEFLECTED POSITION WITH SPHERICAL VALVE IN SHUT-OFF POSITION FOR AFT DUCT
¢ OF VALVE SHAFT
DUCT DIVIDER
ACTUATING LEVER
SPHERICAL VALVE
SECTION
SEALING SPRING SHOWN IN SEALING POSITION
L.E. OF DUCT DIVIDER
FLOW
DIVIDER SEALING SPRING
SPHERICAL VALVE IN SHUT-OFF POSITION FOR AFT DUCT
SPRING TYPE SEAL SHOWN IN DEFLECTED SEAL POSITION
SPHERICAL VALVE IN FLOW THRU POSITION
DIVIDER L.E
DIVIDER SEAL

3




REQ'D PART NO REQ'D
ASSEMBLY OPP.
NEXT ASSY USED ON
APPLICATION QTY REQD

4

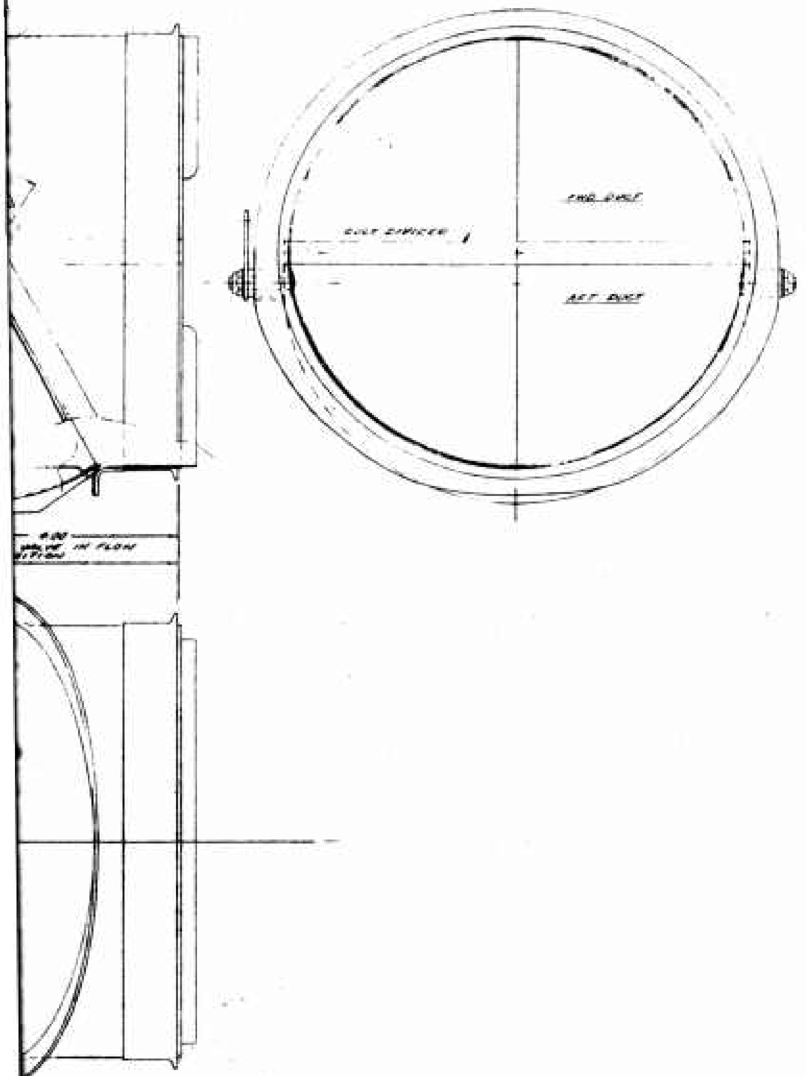
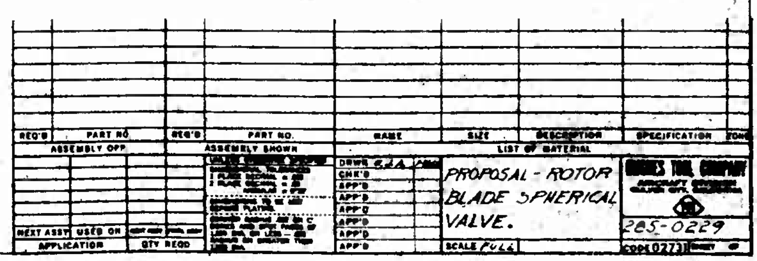
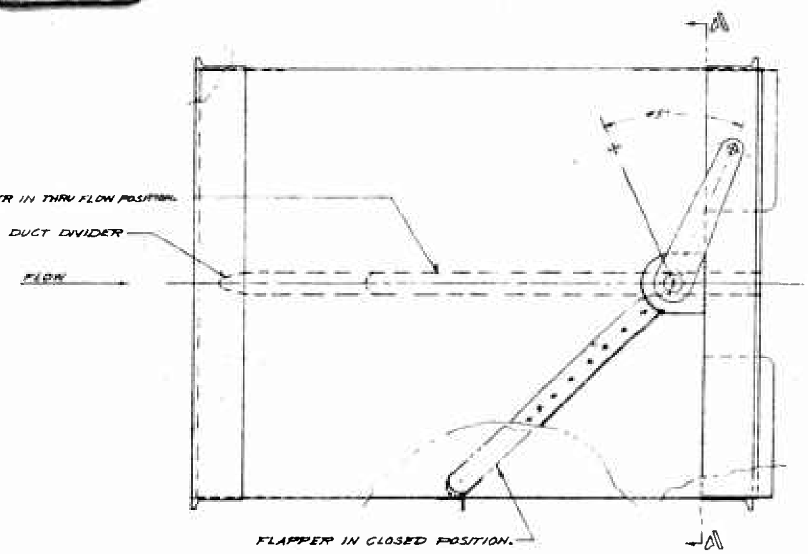

DUCT DIVIDER
FWD DUCT
AFT DUCT
4.00
VALVE IN FLOW

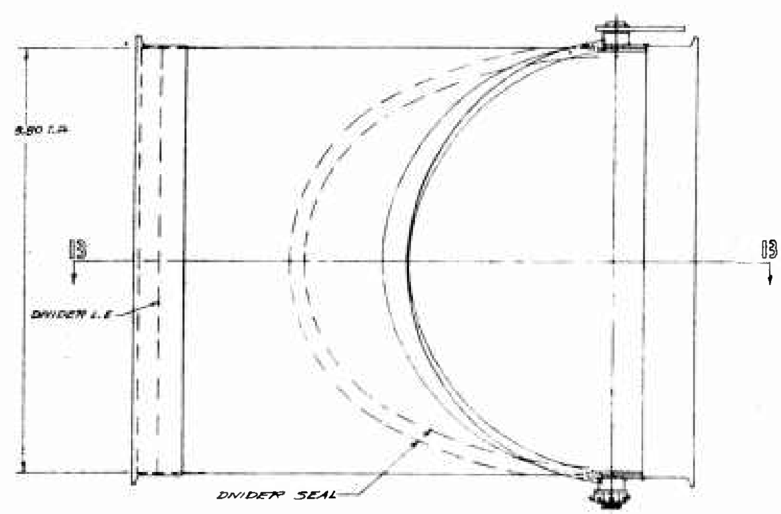
Figure 4-8.


PROPOSAL - ROTOR BLADE SPHERICAL VALVE.
HUGHES TOOL COMPANY
285-0229
SCALE FULL
LIST OF MATERIAL
ASSEMBLY OPP.
ASSEMBLY SHOWN
APPLICATION

1

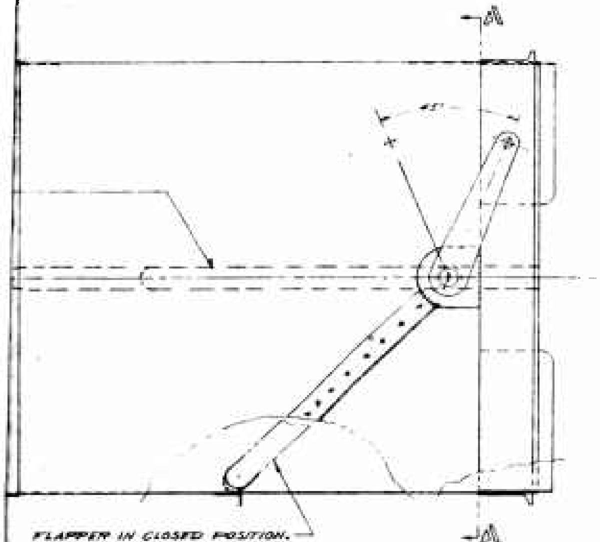


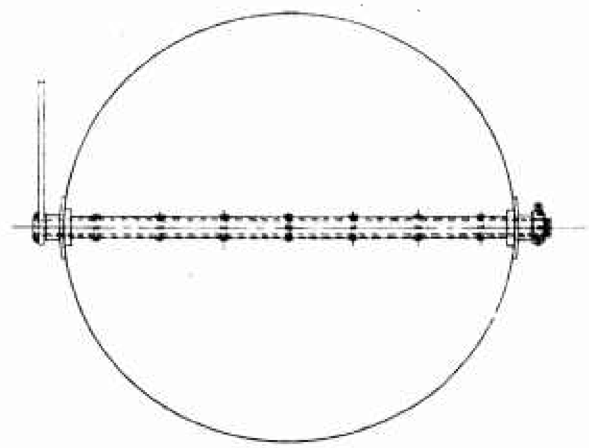
A
FLAPPER IN THRU FLOW POSITION
FIXED DUCT DIVIDER
FLOW
FLAPPER IN CLOSED POSITION.
A

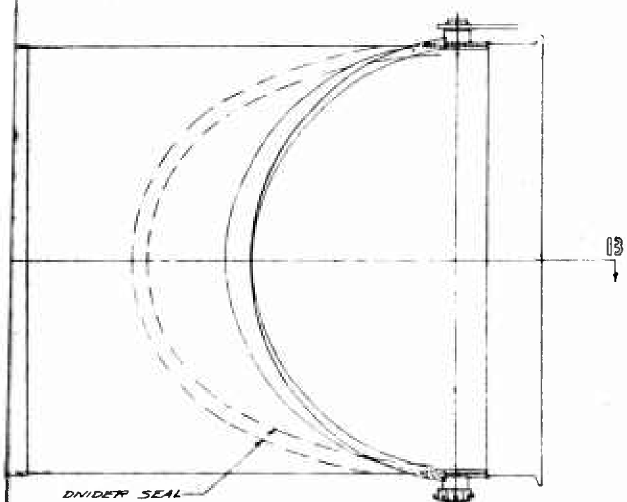
B
B
DIVIDER L.E
DIVIDER SEAL

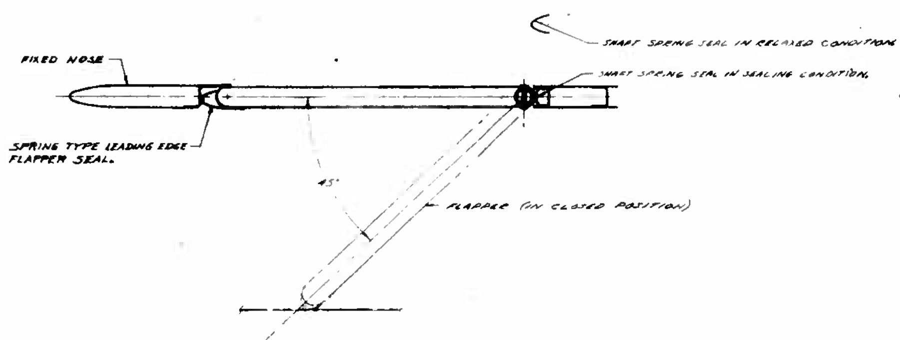
2



SECT. A-A

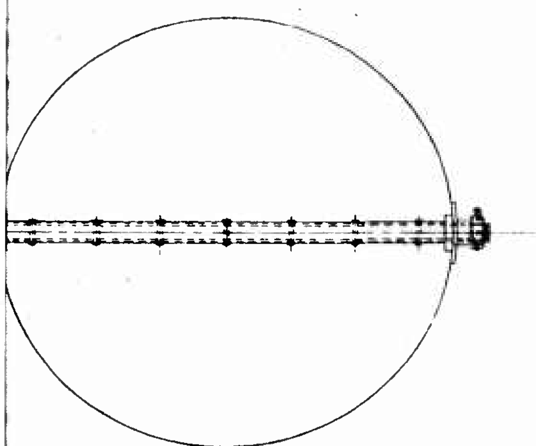


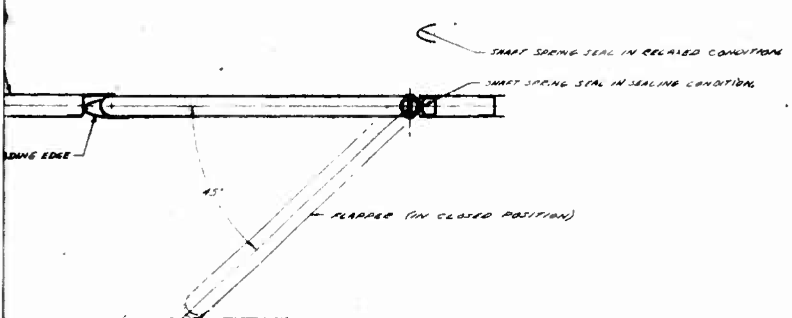

SECT. B-B

NOTES: 1. ALL SPRING TYPE SEALS TO BE OUT OF A HIGH TEMP CORR RESIS. STL., A286, INCONEL X, RENE 41, OR HAYNES 25.

3



SECT. A-A




SECT. B-B

NOTES: 1. ALL SPRING TYPE SEALS TO BE OUT OF A HIGH TEMP. CORR. RESIS. STL., A286, INCONEL X, RENE 41, OR HAYNES 25.

Figure 4-9.

| REQ'D | PART NO. | REQ'D | PART NO. | NAME | SIZE | DESCRIPTION | SPECIFICATION | ZONE |
|---|---|---|---|---|---|---|---|---|
| | | | | | | | | |
| | | | | | | | | |
| | | | | | | | | |
| | | | | | | | | |
| | | | | | | | | |
| | | | | | | | | |

ASSEMBLY OPP. | ASSEMBLY SHOWN | LIST OF MATERIAL

NEXT ASSY | USED ON — APPLICATION | QTY REQD

UNLESS OTHERWISE SPECIFIED

DRWN E. J. A. | CHK'D | APP'D | APP'D | APP'D | APP'D | APP'D | APP'D

PROPOSAL - ROTOR BLADE FLAPPER VALVE.

SCALE FULL

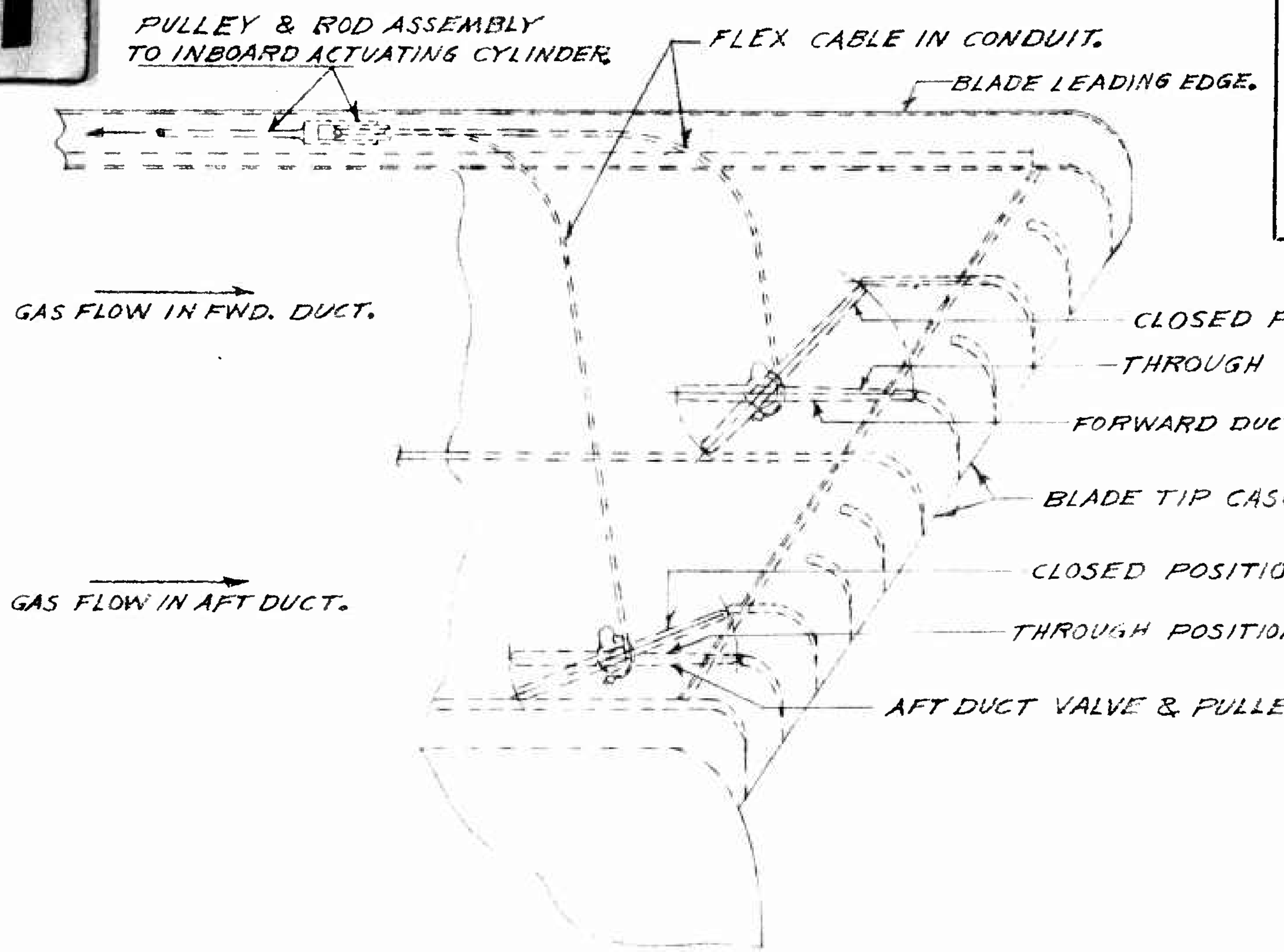
HUGHES TOOL COMPANY AIRCRAFT DIVISION CULVER CITY, CALIFORNIA

285-0230

CODE 02731

1

DO NOT SCALE




| | | | | |
|---|---|---|---|---|
| | | | | |
| REQD | PART NO. | REQD | PART NO. | |
| ASSEMBLY OPP. | | ASSEMBLY SHOWN | | |

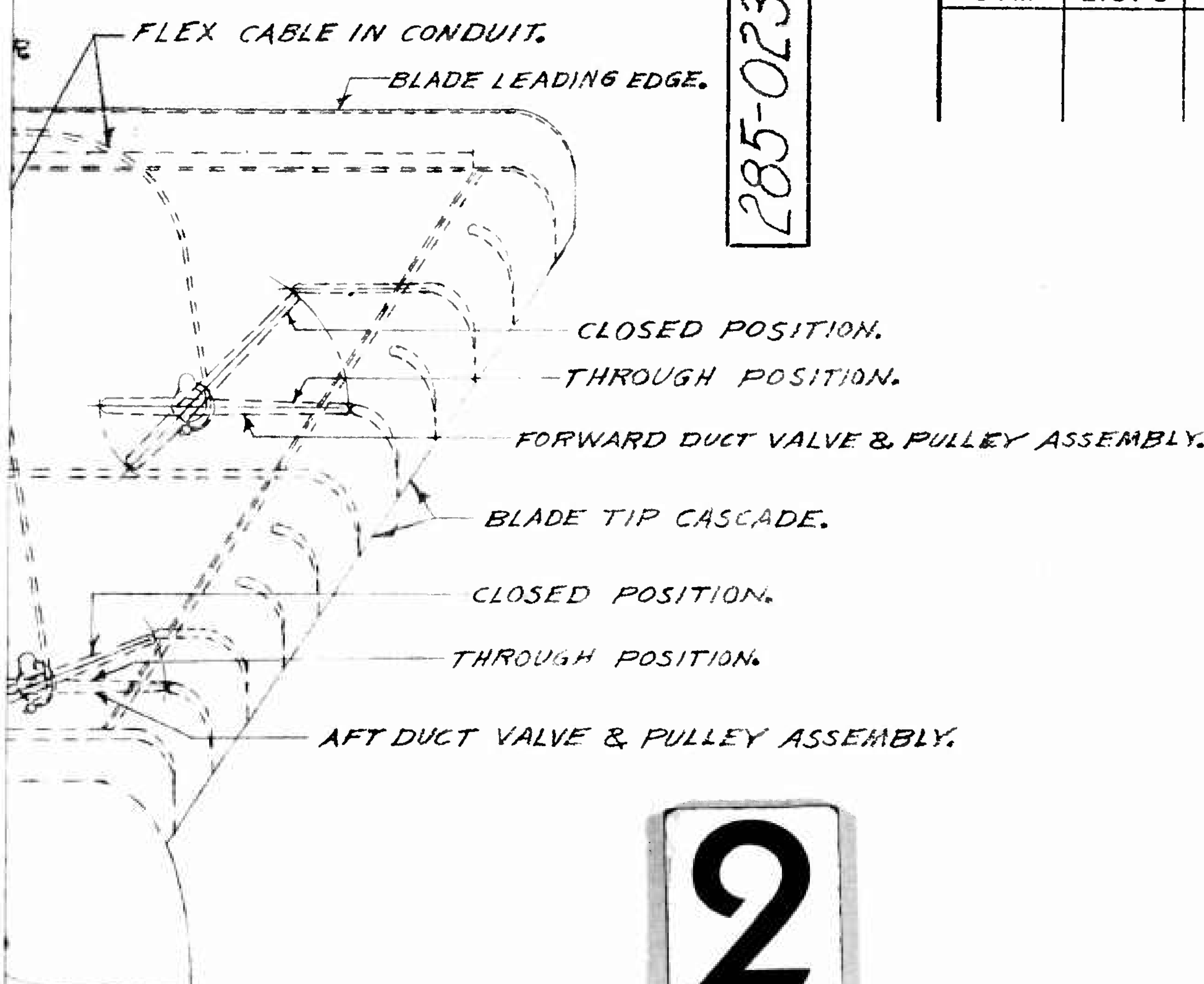
| | | | | | |
|---|---|---|---|---|---|
| | | | | UNLESS OTHERWISE SPECIFIED | DRWN |
| | | | | DIMENSIONAL TOLERANCES<br>3 PLACE DECIMAL ± .010<br>2 PLACE DECIMAL ± .03<br>ANGULAR ± 0°30' | CHK'D<br>APP'D |
| | | | | DIMENSIONS TO BE MET BEFORE PLATING. | APP'D<br>APP'D |
| | | | | CORNER RADIUS .062 ON C' BORES AND SPOT FACES OF 1.250 DIA. OR LESS — .093 RADIUS ON GREATER THAN 1.250 DIA. | APP'D<br>APP'D<br>APP'D |
| NEXT ASSY | USED ON | NEXT ASSY | FINAL ASSY | | |
| APPLICATION | | QTY REQD | | | |

DIETERICH-POST CLEARPRINT 1000H-10

DO NOT SCALE

285-0231

| REVISIONS | | | |
|---|---|---|---|
| SYM | E.O.'S | DESCRIPTION | DRWN |
| | | | |




2

Figure 4-1
Report 285

| | REQD | PART NO. | NAME | SIZE | DESCRIPTION | SP |
|---|---|---|---|---|---|---|
| | | | | | | |
| ASSEMBLY SHOWN | | | LIST OF MATERIAL | | | |

UNLESS OTHERWISE SPECIFIED
DIMENSIONAL TOLERANCES
3 PLACE DECIMAL ± .010
2 PLACE DECIMAL ± .03
ANGULAR ± 0°30'

DIMENSIONS TO BE MET BEFORE PLATING.

CORNER RADIUS .062 ON C' BORES AND SPOT FACES OF 1.250 DIA. OR LESS — .093 RADIUS ON GREATER THAN 1.250 DIA.

| NEXT ASSY | FINAL ASSY |
|---|---|
| QTY REQD | |

| | | |
|---|---|---|
| DRWN | SALLOWS | 3-23-62 |
| CHK'D | | |
| APP'D | [illegible] | 3-23-62 |
| APP'D | | |
| APP'D | | |
| APP'D | | |
| APP'D | | |
| APP'D | | |

SCHEMATIC - BLADE TIP DUCT VALVE SYSTEM.

HUGHES AIRCR
CULVER

285

SCALE

CODE 02

VELLUM "B" SIZE FORM 94

285-0231

| REVISIONS | | | | | |
|---|---|---|---|---|---|
| SYM | E.O.'S | DESCRIPTION | DRWN | APP'D | DATE |
| | | | | | |

3

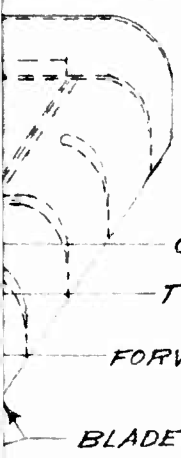



Figure 4-10
Report 285-19 (62-19)

| | | | | |
|---|---|---|---|---|
| | | | | |
| RT NO. | NAME | SIZE | DESCRIPTION | SPECIFICATION |
| SHOWN | LIST OF MATERIAL | | | |

| | | | | |
|---|---|---|---|---|
| RWISE SPECIFIED | DRWN SALLOWS | 3-23-62 | SCHEMATIC-BLADE TIP DUCT VALVE SYSTEM. | HUGHES TOOL COMPANY<br>AIRCRAFT DIVISION<br>CULVER CITY, CALIFORNIA |
| TOLERANCES<br>MAL ± .010<br>MAL ± .03<br>LAR ± 0°30' | CHK'D | | | |
| | APP'D | 3-23-62 | | |
| | APP'D | | | |
| O BE MET<br>NG. | APP'D | | | |
| US .062 ON C'<br>POT FACES OF<br>LESS — .093<br>REATER THAN | APP'D | | | 285-0231 |
| | APP'D | | | |
| | APP'D | | SCALE | CODE 02731 SHEET OF |

VELLUM "B" SIZE FORM 9487 REV. 1-61

its unstable characteristic. However, some type of damper must be connected to the door to slow its motion down as the door moves sideways toward its seat. The high driving force of gas pressure might break something if the door did not have a damper on its motion.

4.4.3 Figure 4-10 (HTC Drawing 285-0231)

This drawing shows the plan view of a blade tip two position valve system which can reduce the tip nozzle area by 50% following failure of one engine. The area change is accomplished by movement of two doors hinged on vertical shafts near their quarter-chord point. For normal two-engine operation, both doors will be oriented parallel to the blade span and will be secured from vibrating by the trailing edge door seal which will act as a spring detent. If an engine fails and the Differential Pressure Sensor sends a signal to operate the tip valves, a rod which is located in closely spaced fairleads within the leading edge fairing will be moved inboard by approximately 1/2 inch. This inboard motion of the rod will pull the loop of the tip-located cable inboard also. The free ends of this cable run through fairleads and over scuff-plates to sectors at the upper and lower ends of the vertical door shafts. All of the cables, scuff plates, fairleads, and sectors are located in the space between the duct wall and the outer blade skin.

When the actuation signal is received, the leading edge rod and tip cable will move inboard, rotating the two doors through an angle until their upstream edge reaches the side of the duct wall toward the blade trailing edge. The door trailing edges will move toward the blade leading edge and will each contact one of the nozzle turning vanes. The moveable doors are equipped at the leading and trailing edge with spring seals, so that when they move to the closed position, they seal off the leading and trailing edges. The upper and lower edges of the moveable doors will have such a small clearance to the duct liner that no appreciable amount of gas will leak past those two edges of the doors.

Figure 4-10 shows that the doors in their closed position will together block off one-half of the originally open number of passages through the tip cascade. The gas from the remaining good engine will pass down both blade ducts, instead of through just one duct, as it would for the designs of Figures 4-8 or 4-9. The one engine flow through the two ducts will experience less pressure drop than the same flow would if it were confined to one duct. About 4% more rotor horsepower will be available for this Figure 4-10 design. The increased "g" loading and more complicated actuation system of the tip-located valve may require more effort to obtain a satisfactory operating valve than one located near the blade root.

It should be noted that the high centrifugal loading just mentioned can be used to make the valve doors return automatically to the "Open" position if the second engine is restored to power and the "engine failure" signal is removed. Sufficient restoring moment can easily be obtained about the door shafts by proper location of mass without resorting to springs to return the doors to the "Open" position.

A summary of the design features of the root and tip blade valves discussed here follows in Table 4-2.

TABLE 4-2

COMPARISON OF BLADE DUCT VALVE DESIGNS

| Figure No. | 4-8 | (Alternate to 4-8; not shown) | 4-9 | 4-10 |
|---|---|---|---|---|
| Door Type | One Ball-Socket | One Flat Door | One Flat Door | Two Flat Doors |
| Valve Location | Blade Root | Blade Root | Blade Root | Blade Tip |
| Hinge Location on Door | Downstream | Upstream | Downstream | Near 1/4 Chord |
| Actuation Force | Low Friction Only | High Friction & Gas Pressure | Low Unstable | Relatively high due to tip "g" loads |
| Pressure Drop In Shutoff Position | High | Low | Low | Low |
| Difficulty of Sealing | Low | High | Low | Low |
| Difficulty of Manufacturing (Relative Cost) | High | Low (If sealing problem solved easily) | Low | Low |

As in the case of alternate diverter valve designs discussed in the preceding section, it is felt that any of the blade valves mentioned above could be used without affecting the results of the Phase I dynamic study. In addition, any of these blade duct valve designs could be used for any of the propulsion arrangements of Figures 4-1a, 4-1b, or 4-1c.

## 4.5 POTENTIAL COMBINATIONS AND FINAL CHOICE OF ENGINE LOCATION, DIVERTER VALVES, AND BLADE DUCT VALVES

It has been pointed out in Figure 4-1 of Paragraph 4.2, that three propulsion arrangements of gas generator and diverter valves are possible. Paragraph 4.3 described four diverter valves that could be used with Figure 4-1a or 4-1b configurations and one diverter valve, the G. E. J85 valve that could be used with the Figure 4-1c propulsion arrangement. In addition, Paragraph 4.4 discussed two blade root duct valves and one blade tip valve to reduce effective nozzle area by 50% in case of a failure of one engine. This represents a total potential combination of:

$\left[(2 \times 4) + (1 \times 1)\right] \times 3 = 27$ over-all arrangements possible with the variations shown here. Very careful design studies of the pertinent combinations are necessary to select the final components for a particular application. It is felt that any of the diverter valves or blade duct valves could be used with acceptable results. However, an optimum selection of components is possible for any application. This Contractor has started work on the design study required by Reference 14. It is expected that the material contained in this study will be used in the Reference 14 study to produce an optimum arrangement of components for some particular application.

## 4.6 BLADE TIP NOZZLE FLOW CHARACTERISTICS

Prior to conducting rotating blade performance and endurance tests, some preliminary calibration runs were made to check the flow characteristics of the blade duct and tip nozzle cascades. These tests are reported in Reference 15. The tests were conducted with the rotor restrained so that it could not turn. Gas flow was directed through one blade only, the other two being capped off. Mass flow was measured at a measuring station below the rotor rotating seal. Blade tip thrust was measured by the strain in a calibrated link that reacted blade thrust.

When these tether test data were examined, they were found to bear out the original assumptions with the exception of the tip nozzle flow coefficient, $C_W$. The apparent value of $C_W$ from the static tether tests of Reference 15 is shown in Figure 4-11. The value of $C_W$ is greater than

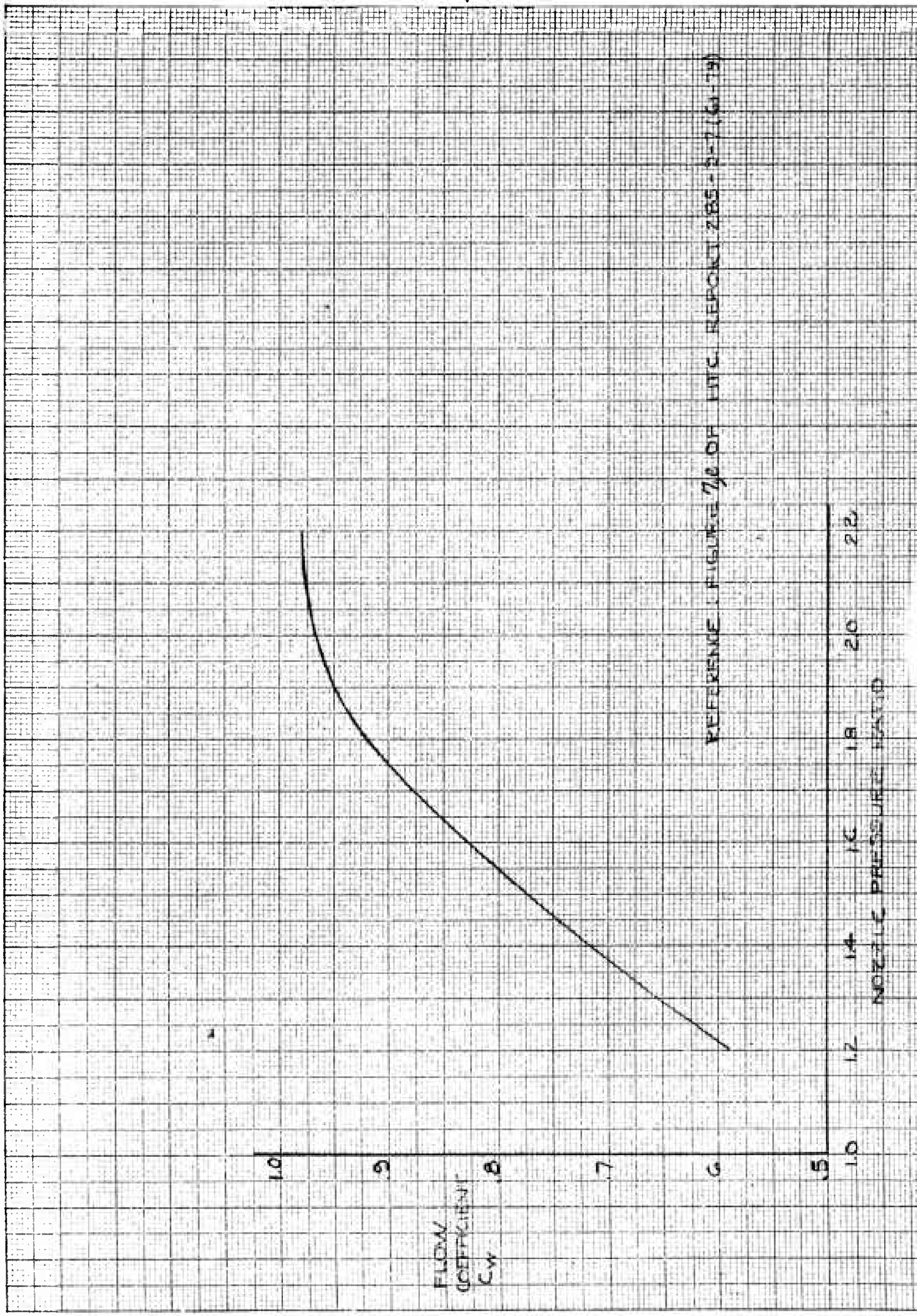



Figure 4-11. Flow Coefficient Versus Nozzle Pressure Ratio

0.95 for all nozzle pressure ratios above 1.9 (approximately choking pressure ratio), but falls gradually down to about 0.59 at pressure ratios of 1.20. This value of flow coefficient is far below the value that can be expected for this pressure ratio for ordinary nozzles. Figure 18-4 of Reference 16 gives values of flow coefficients of 0.85 or higher at a pressure ratio of 1.2. Therefore, according to the tether tests of Reference 15, the flow coefficient of the hot cycle blade apparently deteriorates drastically at low pressure ratios such as are found at engine idle.

The effect of this apparently low flow coefficient on the mass flow function which will pass through the nozzles is shown as Curve A in Figure 4-12 as a function of nozzle pressure ratio. Also shown in Figure 4-12 is Curve B which is the "ideal" flow through the same nozzle with a conventional variation of flow coefficient, such as found in Reference 16. In addition, Curve C is given which gives the mass flow function as determined from later tests of the rotor with the blades turning.

It can be seen in Figure 4-12 that the measured flow function for the rotating blade case is almost identical to the flow function of the "ideal" case, indicating that there is no discrepancy in flow coefficient when the blade is rotating. Yet, at pressure ratio of 1.2, the flow function, using flow coefficient as determined in tether tests, is only about half of that for Curve B or C.

The explanation and indicated solution of this discrepancy is found in an interrelationship between internal and external flow. If the blade nozzle is located in a spot where the external pressure produces a pressure below atmospheric, the actual nozzle pressure ratio will be increased substantially. As a result, the nozzle pressure ratio will increase toward choking value, and the working value of flow coefficient will be the conventional high value shown at the right end of Figure 4-11.

Such conditions are quite probable for the nozzle installed at the tip of a rotating blade. Highly negative pressures may exist at the core of the tip vortex very close to the exit of the nozzle. In this condition, the rotating nozzle pressure ratio becomes higher than for a nozzle discharging to atmospheric pressure. According to Reference 17, the negative pressures prevailing in the core of a tip vortex may be up to

$$\frac{\Delta P}{q} \simeq -3.$$

The effect of this negative pressure due to the external flow field on the nozzle pressure ratio was computed at 170 RPM for two runs remote

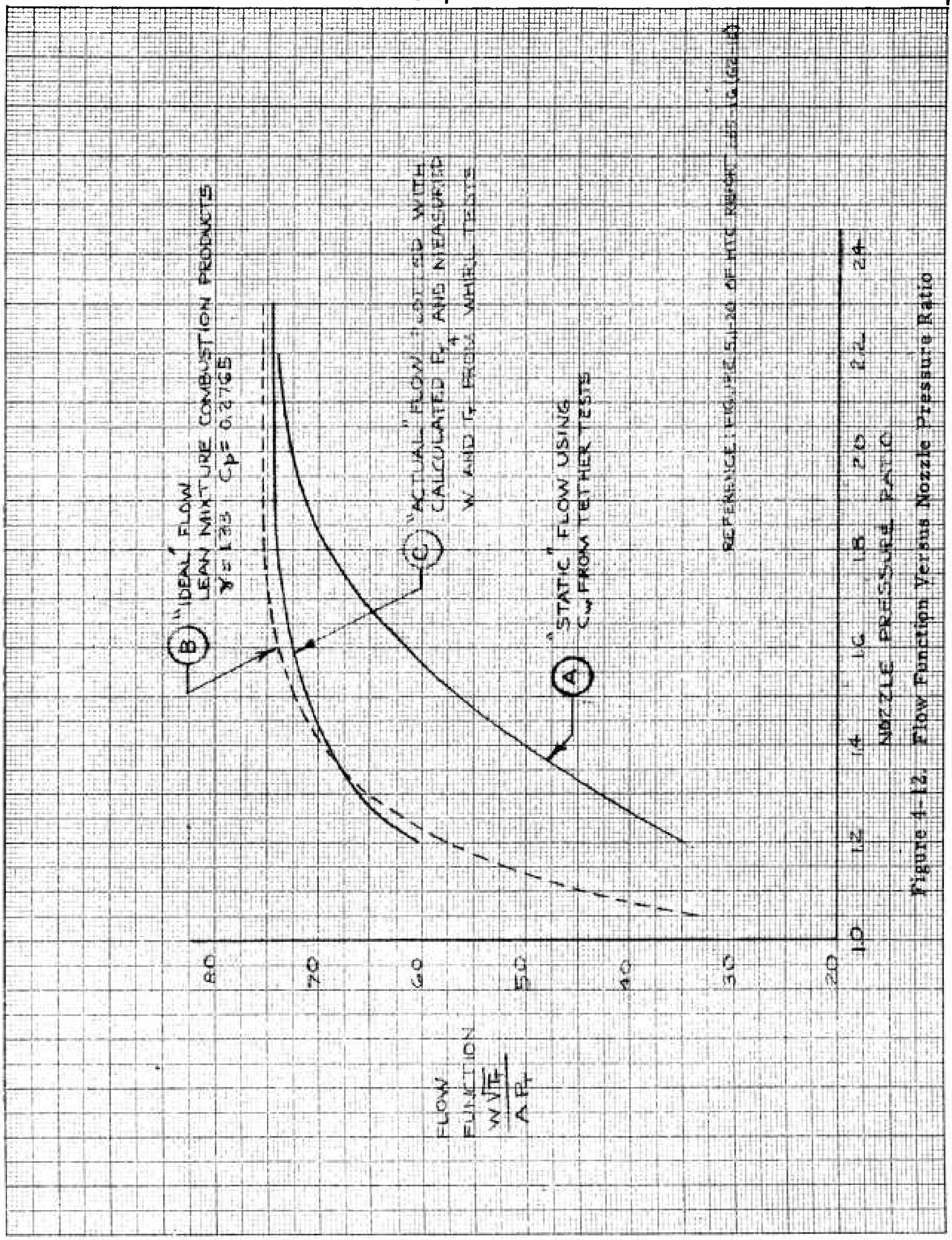



Figure 4-12. Flow Function Versus Nozzle Pressure Ratio

from choking. It was found that with

$$\frac{\Delta P}{q}$$

equal to only -2.0, nozzle pressure ratio would increase from 1.4 to about 1.9, or greater than choking. The remaining runs would need much less negative pressure to shift nozzle pressure ratio enough that the flow coefficient might reach its maximum value.

The practical application of the information contained in this Section is as follows: When the rotor is at rest, the static flow coefficient of the tip nozzles is so low that the nozzles will pass only about 50% of the flow function which the engines will want to put out if they are to run normally. If the engines do not discharge this gas, they will surge badly. Therefore, it is necessary to expose the engines to more effective nozzle area when the rotor is not turning than the blade nozzles alone offer. This will be accomplished by moving the blade diverter valves to about 25-30% of the travel from the "overboard" position. The diverter valve nozzles will have conventional flow coefficients for the overboard position. Consequently, they can accept the full engine flow with no surging. By opening the diverter valves a little bit, some gas will go to the rotor and all the rest will go out the "overboard" nozzle with no trouble. With the power control levers advanced slightly from idle, the gas that does go to the rotor will provide enough power to drive the rotor in low pitch up to 70% speed, or 170 RPM. At this value of rotor RPM, the

$$\frac{\Delta P}{q}$$

of external flow field will be sufficient to provide a $-\Delta P$ which will combine with the internal pressure to raise even the near-idle nozzle pressure of perhaps 1.2 to an effective nozzle pressure ratio near choke. The operating nozzle flow coefficient will then be near the maximum value of 0.96, instead of down to about 0.59. With this condition, the tip nozzle can finally accept the full engine flow function and will not cause the engine to surge. Therefore, when the rotor speed is 170 RPM or higher, the diverter valves can be turned completely to the rotor direction. The complete description of valve operation given in the next section will reflect the two-step diverter valve operation discussed here.

## 4.7 DISCUSSION OF ENGINE AND VALVE OPERATION

The main components of the propulsion system for the hot cycle helicopter configurations will consist of two gas generators, two diverter valves in the fuselage, and three blade duct valves. All of these components must be properly coordinated during engine start, rotor acceleration, normal two engine operation, and one engine out operation.

The pilot will be required to operate and synchronize the following controls from the cockpit for control of power:

(1) Collective pitch stick

(2) Two power control levers (one for each engine)

(3) Two diverter valve controls

In addition to these pilot-operated controls, there will be automatic operation of a blade duct valve in each blade to reduce effective tip nozzle area when one engine quits or is shutoff to simulate an engine failure. This automatic operation is initiated whenever a signal is developed which indicates one engine has failed. This engine failure signal is produced by a device called a differential pressure sensor. Figures 4-13a through 4-13i show schematic drawings of the general relationship of engines, levers, controls, and valves to each other. The differential pressure sensor is shown located in the Figure 4-13 schematics between the engines.

The operation of the differential pressure sensor is based on recognition of the fact that with a fixed nozzle area, the power level of a jet engine is measured directly by the total pressure of the gases discharged from the engine. Further, if two engines of equal quality are operated at the same power control setting, they will produce the same total pressure. Therefore, if a total pressure pickup is led from each engine to opposite sides of a sealed chamber which is split by a diaphragm, the diaphragm will sense no signal; i.e., will not deflect if two equal quality engines are operated at the same power setting. Conversely, if either of those two equal quality engines is malfunctioning, the engines will not develop equal total pressure and the diaphragm will deflect under the unequal pressure. The motion of the diaphragm can be used to open or close switches to command the diverter valves and blade duct valves to move to the proper position for whatever conditions prevail. If the engines are not of equal quality, and to prevent inadvertant shutoff of a good engine, a value of differential pressure will be selected which must be exceeded before the automatic motion of the valves will be initiated.

The differential pressure sensor shown throughout Figure 4-13 has its total head tubes located immediately downstream from the gas generator turbines. This location is removed by several feet from the area in the region of the rotating seal where static pressures of the flow from the two engines will be brought to the same value. There are a number of struts and baffles in the path from the engine to the plenum at the rotating seal (and above) which will permit a differential total pressure to exist at the back of the engines, even though there is a common static and total pressure above the rotating seal.

The differential total pressure sensor is also shown in Figure 4-14, which is a complete schematic of the over-all hot cycle control system, including all components, such as the primary hydraulic system which operates the diverter valves, etc.. A complete description of the operation of these components is given in Appendix I, which shows in detail how the individual components are made to move. The purpose of this discussion here of Figures 4-13a through 4-13i is to show the proper position of control levers and valves for each major power and rotor condition. The schematics of Figure 4-13 are grouped in accordance with four distinct phases of operation as discussed below.

4.7.1 Start Engines

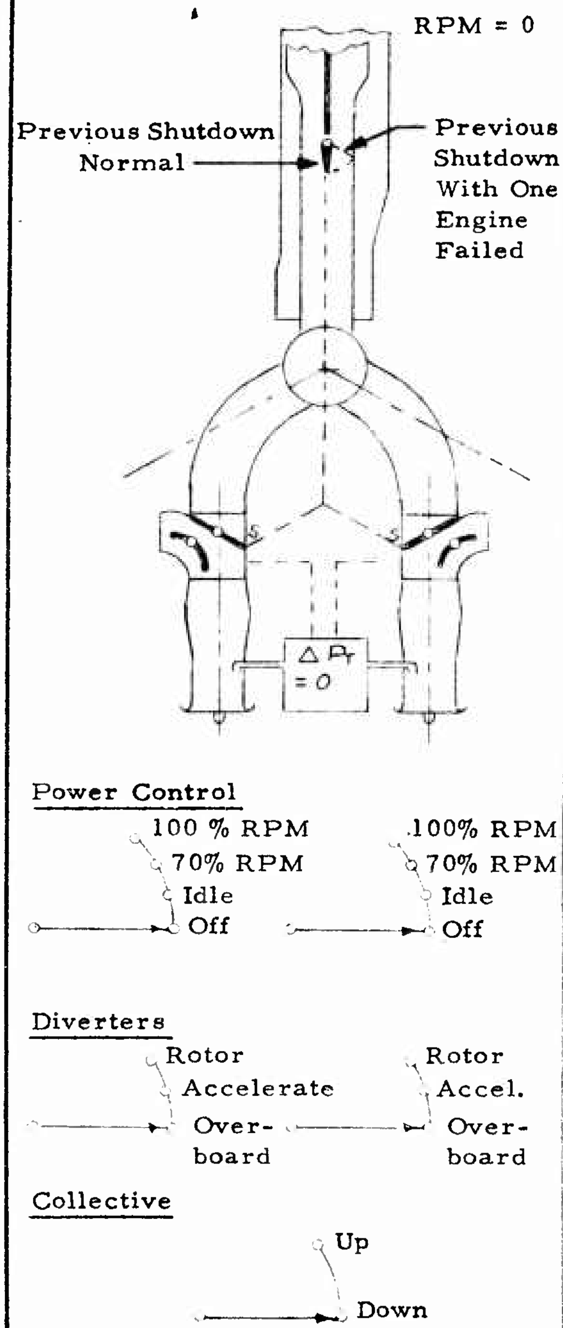
4.7.1.1 Figure 4-13a: Engines Off

| | |
|---|---|
| Collective Stick: | Full down |
| Diverter Valve Controls: | Both set to "Overboard" position |
| Power Control Levers: | Both set to "Off" (See Reference 9 and Section 3.2.1). |
| Diverter Valves: | Both seated in "Overboard" position. |
| Blade Duct Valves: | Valves neutral if previous shut down normal. Valve closed if previous shut down with one engine failed. |
| Rotor RPM: | 0 |

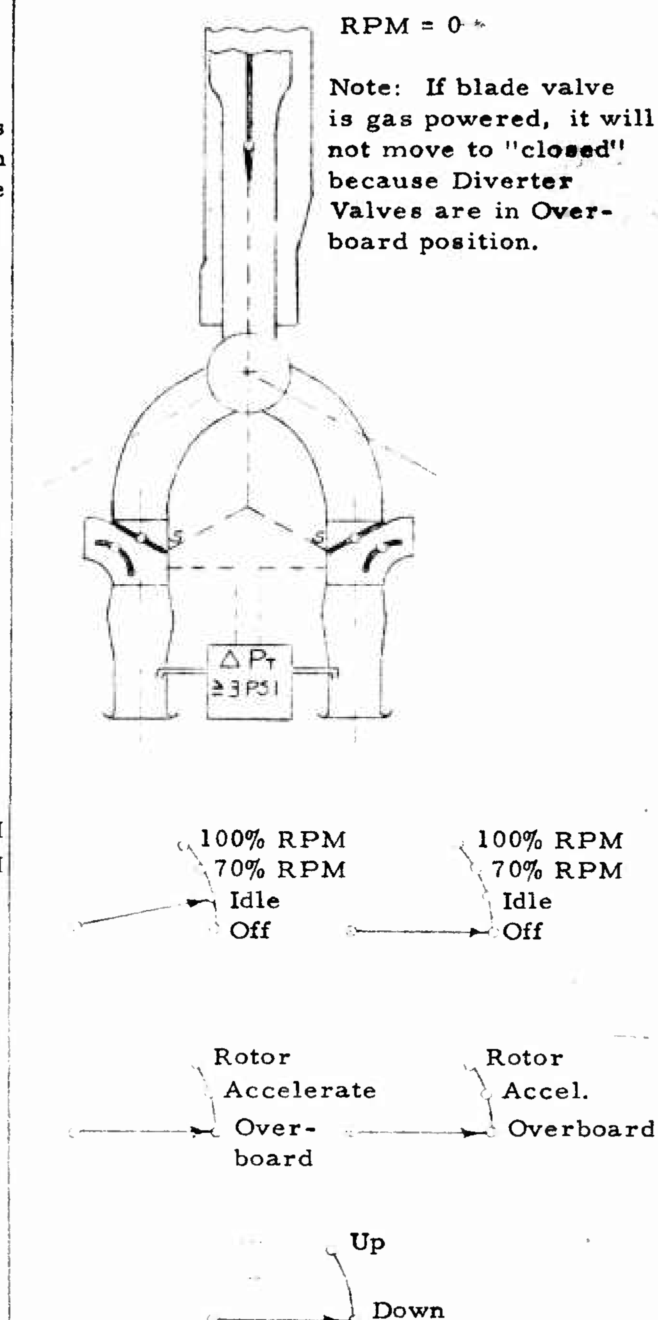
4.7.1.2 Figure 4-13b: Start First Engine

| | |
|---|---|
| Collective Stick: | Full down |

ANALYSIS
PREPARED BY
CHECKED BY




(4-13a) Engines Off




(4-13b) Start First Engine

Figures 4-13 a, b, c, d, e, f, g, h, i - General Arrangement of Gas Generators, Valves, and Controls

| | |
|---|---|
| Diverter Valve Controls: | Both set to "Overboard" position. |
| Power Control Levers: | No. 1 moved to "Idle." No. 2 still "OFF." |
| Diverter Valves: | Both seated in "Overboard" position. |
| Blade Duct Valves: | Neutral |
| Rotor RPM: | 0 |

Note: When the first engine is started, the differential pressure sensor will register a pressure signal of about 3 psi, which is gage pressure of an engine at Idle at seal level. This signal might be expected to operate the blade duct valve unnecessarily every time starting procedure is followed. Such is not the case, as pointed out in Appendix I and Figure 4-14. When the diverter valve controls are in the "Overboard" or "Accelerate" position, there is no electrical circuit to the rotor to actuate the blade duct valve.

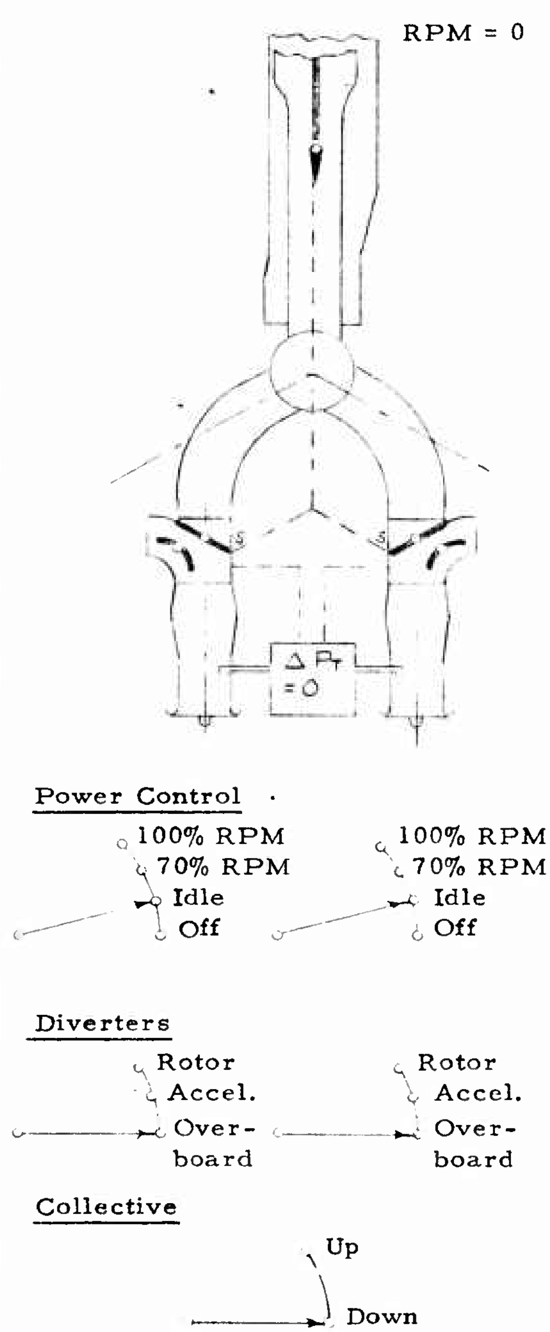
4.7.1.3 Figure 4-13c: Start Second Engine

| | |
|---|---|
| Collective Stick: | Full Down |
| Diverter Valve Controls: | Both set to "Overboard" position |
| Power Control Levers: | Both on "IDLE." |
| Diverter Valves: | Both seated in "Overboard" position. |
| Blade Duct Valves: | Neutral |
| Rotor RPM: | 0 |

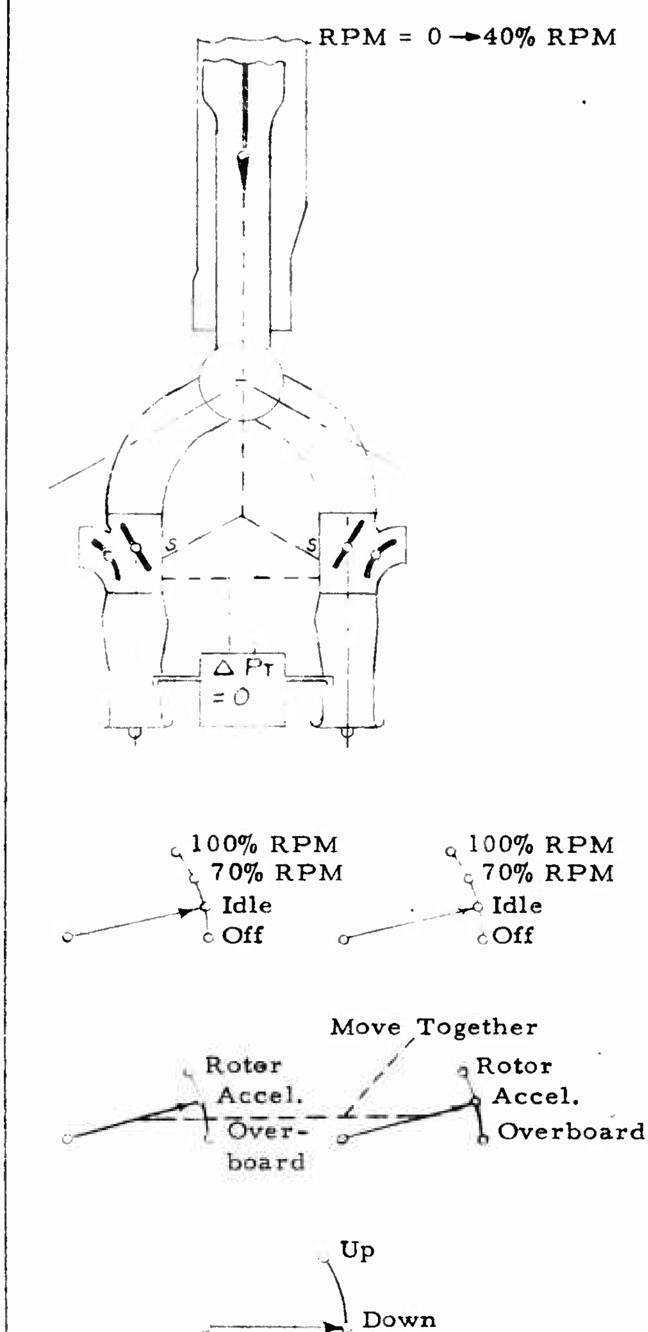
4.7.2 Accelerate Rotor to 100% RPM

4.7.2.1 Figure 4-13d: Move Diverter Valve Controls to "Accelerate"

| | |
|---|---|
| Collective Stick: | Full Down |
| Diverter Valve Controls: | Move both together to "Accelerate" position. |
| Power Control Levers: | Both on "IDLE." |




(4-13c) Start Second Engine




(4-13d) Accelerate Rotor to 100% RPM -- Move Diverters to "Accelerate"

Diverter Valves: Both moved to "Accelerate" position.

Blade Duct Valves: Neutral

Rotor RPM: 0→40%.

Note: This step moves diverter valves to intermediate position which provides sufficient total effective nozzle area to keep the engine on its proper operating line while permitting some gas to go to rotor. This intermediate position of the diverter valves provides the solution to the deficiencies of the nozzle flow coefficient, as discussed in Paragraph 4.6. With the engines at IDLE and collective stick FULL DOWN, it is expected that the rotor will accelerate to about 40% of maximum RPM. As pointed out in Paragraph 4.6, the rotor must accelerate to 70%, or 170 RPM, before it is safe to move diverter valves fully to the rotor position.

4.7.2.2 Figure 4-13e: Move Power Controls to 70% RPM

Collective Stick: Full Down

Diverter Valve Controls: Both on ACCELERATE

Power Control Levers: Move both to 70% RPM (rotor)

Diverter Valves: Both in "Accelerate" position.

Blade Duct Valves: Neutral

Rotor RPM: 40%→70%

Note: This step brings rotor RPM up to value where it is safe to open diverter valves to "Rotor" position.

4.7.2.3 Figure 4-13f: Move Diverter Valves to "Rotor"

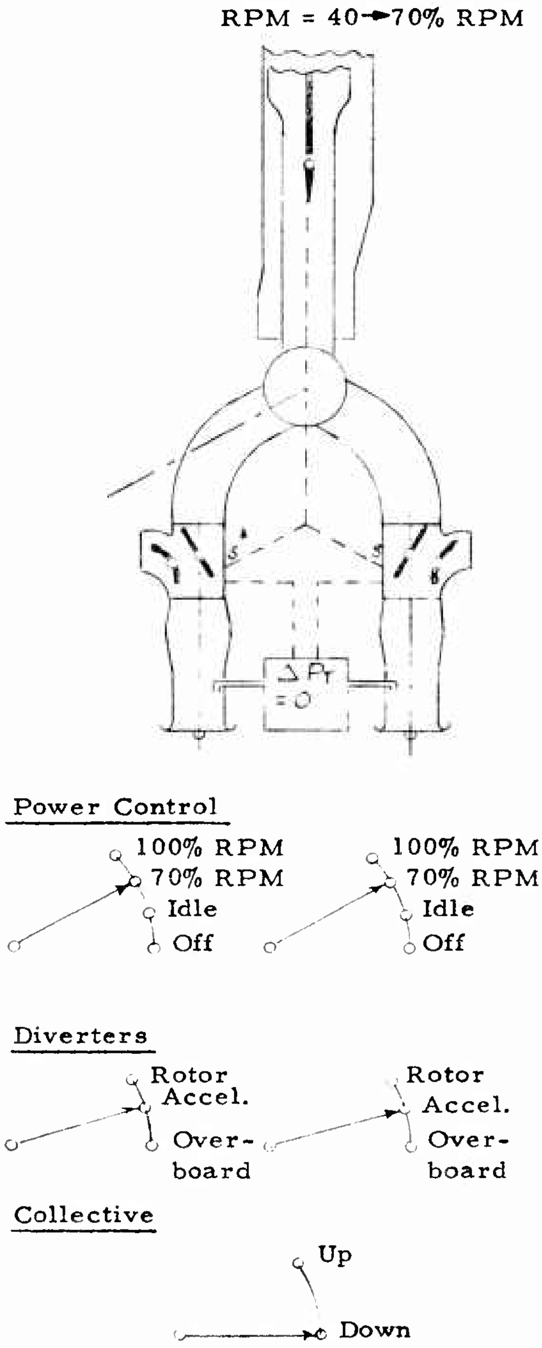
Collective Stick: Full Down

Diverter Valve Controls: Move both to "Rotor" position.

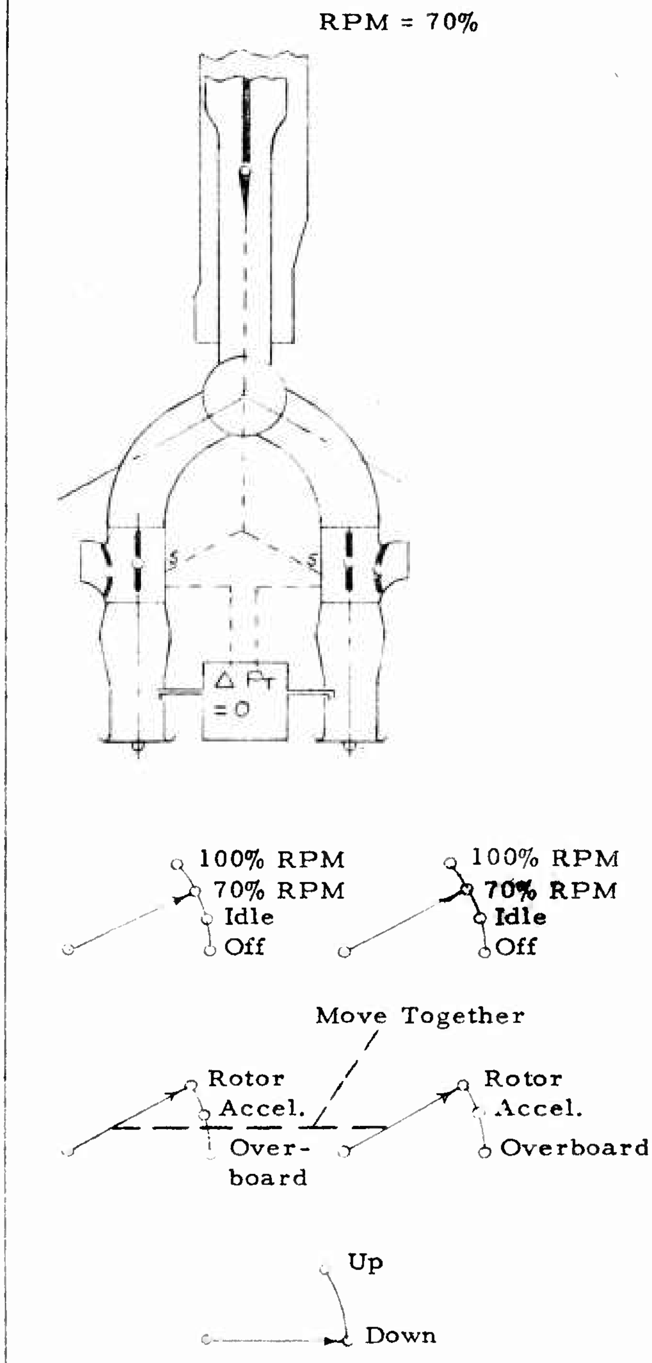
Power Control Levers: Both on "70% RPM" position

Diverter Valves: Both in "Rotor" position.

Blade Duct Valves: Neutral




(4-13e) Accelerate Rotor To 100% RPM -- Move Power Control to 70% RPM




(4-13f) Accelerate Rotor To 100% RPM -- Move Diverter Valves To "Rotor"

| | |
|---|---|
| Rotor RPM: | 70% |

4.7.2.4 Figure 4-13g: Move Power Control to 100% RPM

| | |
|---|---|
| Collective Stick: | Full Down |
| Diverter Valve Controls: | Both on "Rotor" position. |
| Power Control Levers: | Move both together to "100% RPM" |
| Diverter Valves: | Both in "Rotor" position |
| Blade Duct Valves: | Neutral |
| Rotor RPM: | 70%→100% |

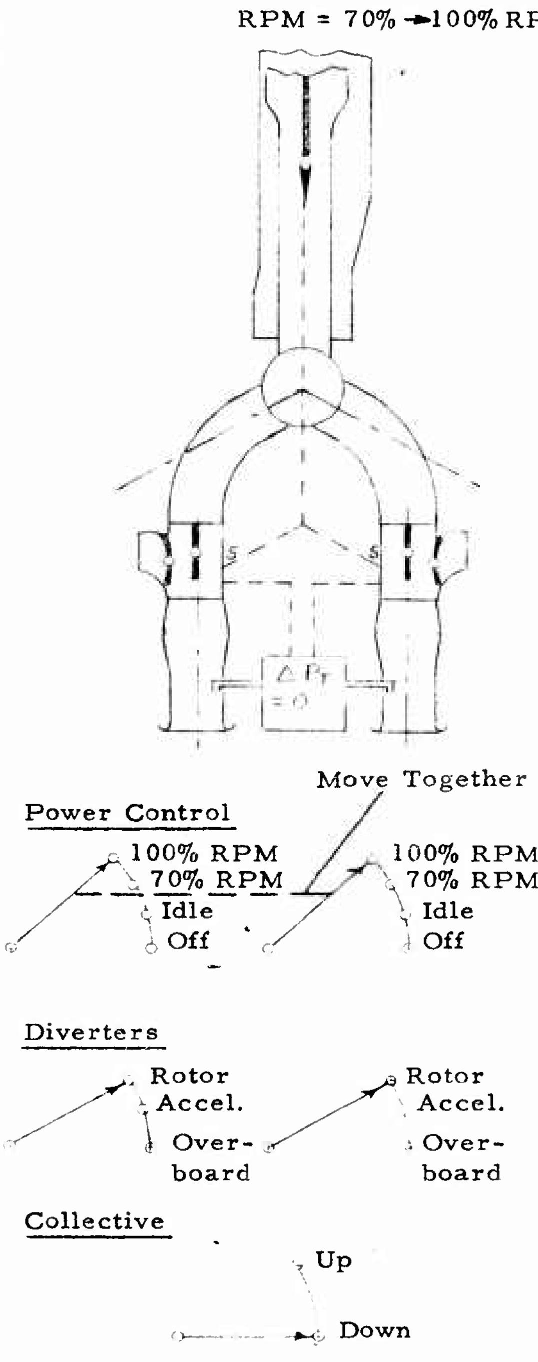
Note: This step completes the "Rotor Acceleration" procedure. Normal powered flight is now possible.

4.7.3 Normal Two Engine Operation

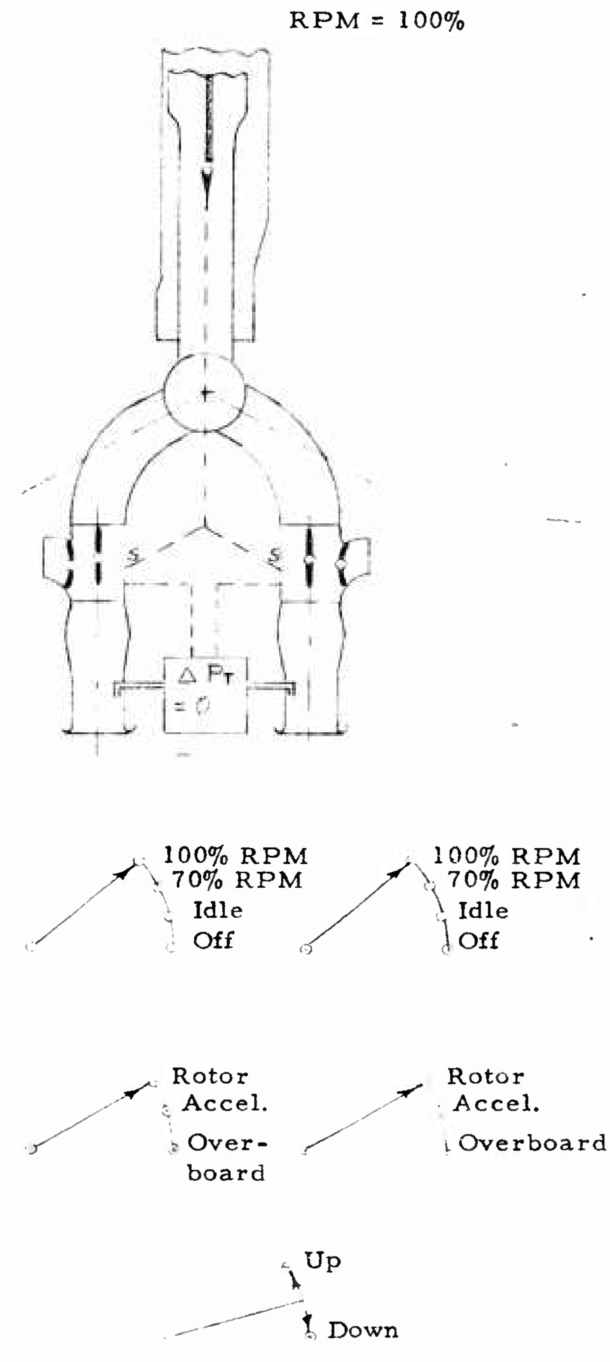
4.7.3.1 Figure 4-13h: Two Engine Operation

| | |
|---|---|
| Collective Stick: | Full down to full up |
| Diverter Valve Controls: | Both in "Rotor" position. |
| Power Control Levers: | Both in "100% RPM" position. |
| Diverter Valves: | Both in "Rotor" position |
| Blade Duct Valves: | Neutral |
| Rotor RPM: | 100% |

Note: At this stage, rotor is up to speed and its speed is governed by the fuel control. The pilot may fly with any collective pitch from full down to full up, and rotor speed will stay essentially constant. The load signal shaft will be properly coordinated with the collective stick to compensate for droop.




(4-13g) Accelerate Rotor to 100% RPM -- Move Power Control to 100% RPM




(4-13h) Two Engine Operation -- 100% RPM -- Collective Pitch Full Down → Up

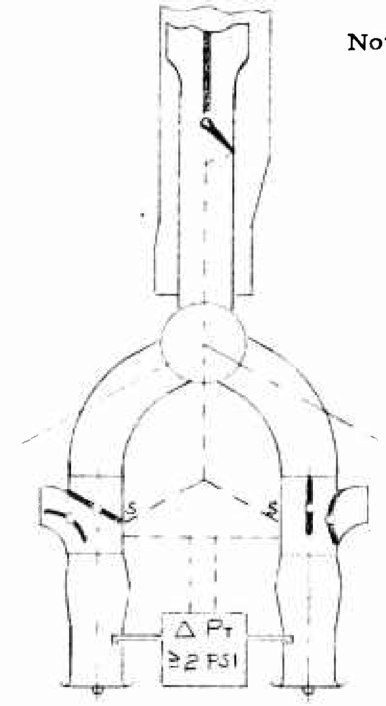
## 4.7.4 Emergency One Engine Out

### 4.7.4.1 Figure 4-13: One Engine Out Operation

| | |
|---|---|
| Collective Stick: | Any position from full down to 50% power position. |
| Diverter Valve Controls: | Both on "Rotor" position. |
| Power Control Levers: | Both on 100% RPM position. |
| Diverter Valves: | Diverter valves for good engine will stay on "rotor" position. If differential pressure sensor senses a $\Delta P_T$ of 2 psi or greater, the diverter valve on the bad engine will be moved to "overboard" position by mechanism shown in Figure 4-14. When diverter valve seats, it will strike microswitch "S." |
| Blade Duct Valves: | Will close after receiving signal which begins when diverter valve seats and hits switch "S." |
| Rotor RPM: | 100% |

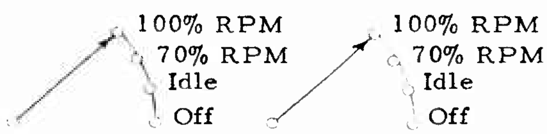
Note: This step describes the automatic operation of the diverter valve of the bad engine and all blade duct valves after one engine fails. It should be noted that the blade duct valve is not in neutral, and will stay there even after the good engine is shut down after landing. When the bad engine is started up again after repair (or returned to power the rotor if a practice failure was used), the blade duct valves will be returned to neutral automatically. This sequence is explained in Appendix I.

RPM = 100%

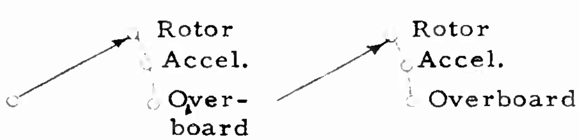



Note: If $\Delta P_t \geq$ 2 psi or more, diverter valve on bad engine will be moved to the "Overboard" position by $\Delta P_t$ sensor, and will strike microswitch "S" as diverter seats. Blade valves will then be energized to close off one duct in each blade. When engines are re-started on ground, blade valves will open after steps 4-13a through 4-13d.

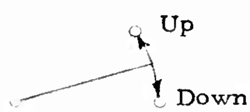
Power Control




Diverters

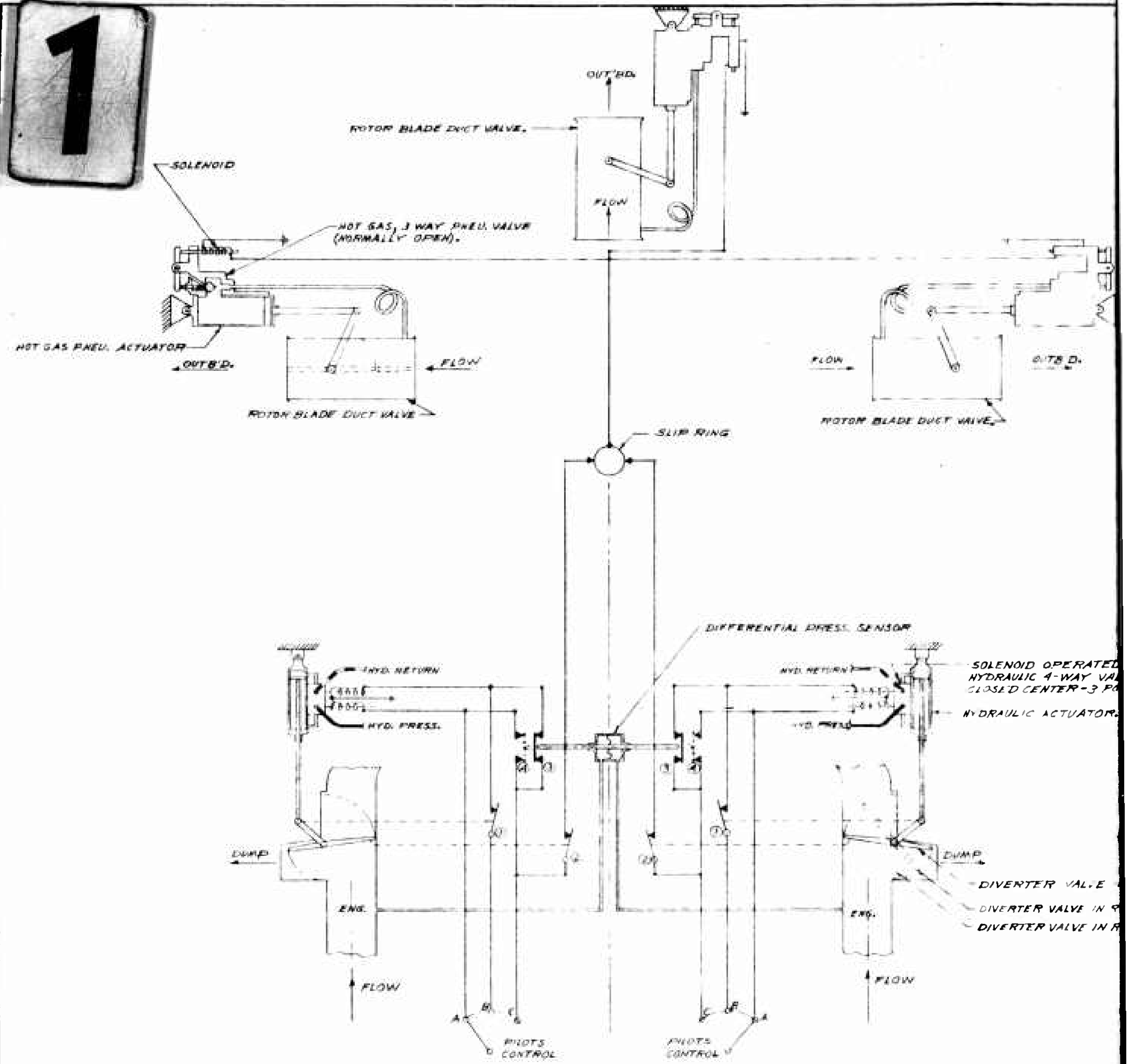



Collective




(4-13i) Emergency One Engine Duct

1




PILOTS CONTROL POSITIONS:

A - DIVERTED POSITION.
B - ROTOR ACCELERATION POSITION.
C - ROTOR FULL FLOW POSITION.

1, 2, 3, 4
MICRO SWITCH POSITIONS

1 OPEN AT ACCEL. POSITION OF DIVERTER VALVE. CLOSED AT ALL OTHERS.
2 CLOSED AT DIVERTED POSITION OF DIVERTER VALVE. OPEN AT ALL OTHERS.
3 CLOSED WHEN $\Delta P_t = 0$ BETWEEN THE TWO ENGINES. OPEN WHEN $\Delta P_t$ = A PRE-DETERMINED VALUE.
4 OPEN WHEN 3 IS CLOSED. CLOSED WHEN 3 IS OPEN.

| REVISIONS | | |
|---|---|---|
| SYM | E.O.'S | DESCRIPTION |
| | | |

2

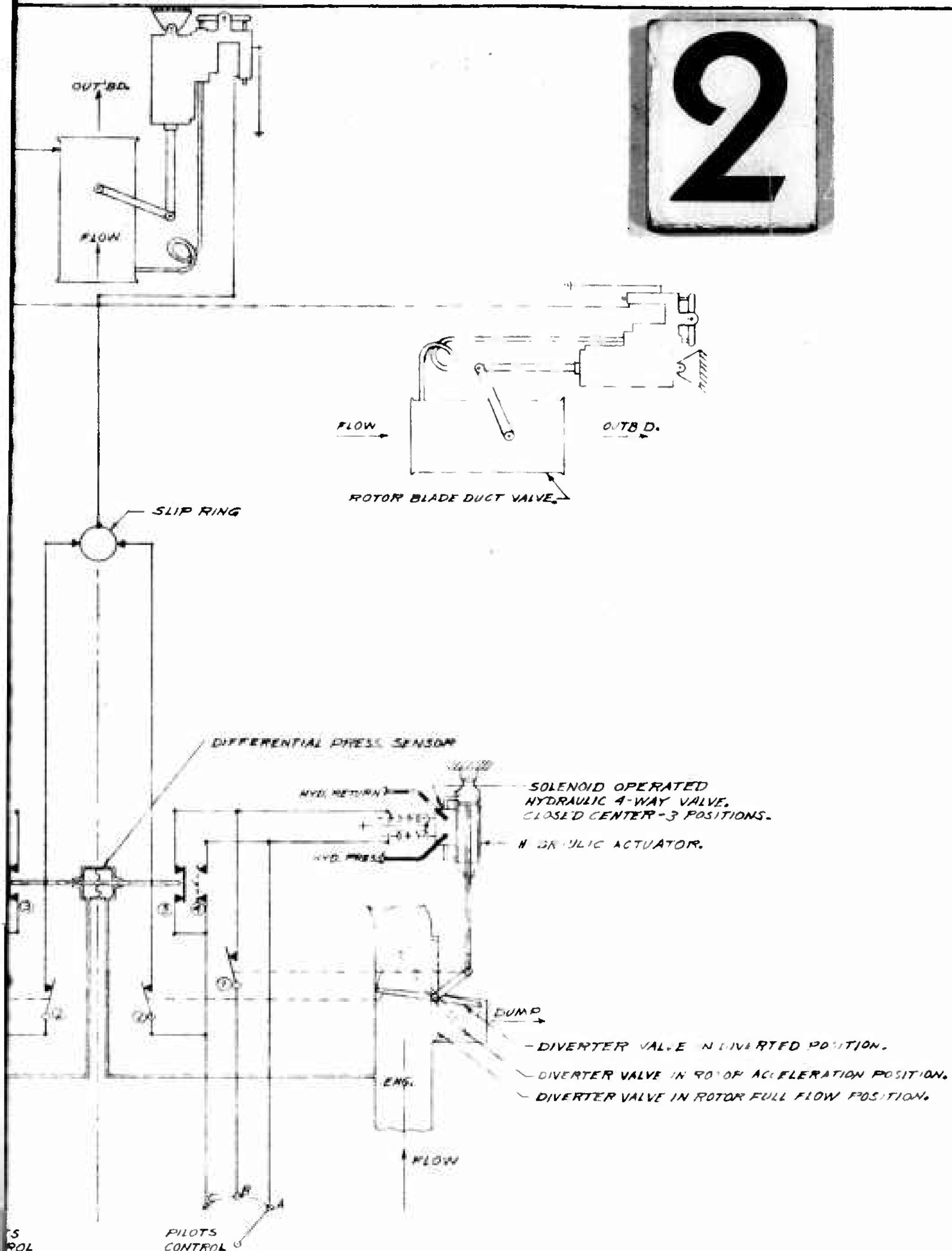



Figure 4-14
Report 285-19 (62

①②③④
MICRO SWITCH POSITIONS

N POSITION.
OSITION.

① OPEN AT ACCEL. POSITION OF DIVERTER VALVE. CLOSED AT ALL OTHERS.
② CLOSED AT DIVERTED POSITION OF DIVERTER VALVE. OPEN AT ALL OTHERS.
③ CLOSED WHEN ΔP = 0 BETWEEN THE TWO ENGINES. OPEN WHEN ΔP = A PRE-DETERMINED VALUE.
④ OPEN WHEN ③ IS CLOSED. CLOSED WHEN ③ IS OPEN.

| REQD | PART NO. | REQD | PART NO. | NAME | SIZE | DESCRIPTI |
|---|---|---|---|---|---|---|
| ASSEMBLY OPP. | | ASSEMBLY SHOWN | | LIST OF MATERIAL | | |

| NEXT ASSY | USED ON | NEXT ASSY | FINAL ASSY | UNLESS OTHERWISE SPECIFIED | | | |
|---|---|---|---|---|---|---|---|
| APPLICATION | | QTY REQD | | DIMENSIONAL TOLERANCES 3 PLACE DECIMAL ± .010 2 PLACE DECIMAL ± .03 ANGULAR ± 0°30' DIMENSIONS TO BE MET BEFORE PLATING. CORNER RADIUS .062 ON C' BORES AND SPOT FACES OF 1.250 DIA. OR LESS — .093 RADIUS ON GREATER THAN 1.250 DIA. | DRWN Sallows 2/15/62 CHK'D APP'D 2/16/62 APP'D APP'D APP'D APP'D APP'D | SCHEMATIC – H CYCLE GAS CONT SYSTEM. | SCALE — |

| REVISIONS | | | | | |
|---|---|---|---|---|---|
| SYM | E.O.'S | DESCRIPTION | DRWN | APP'D | DATE |
| | | | | | |

3

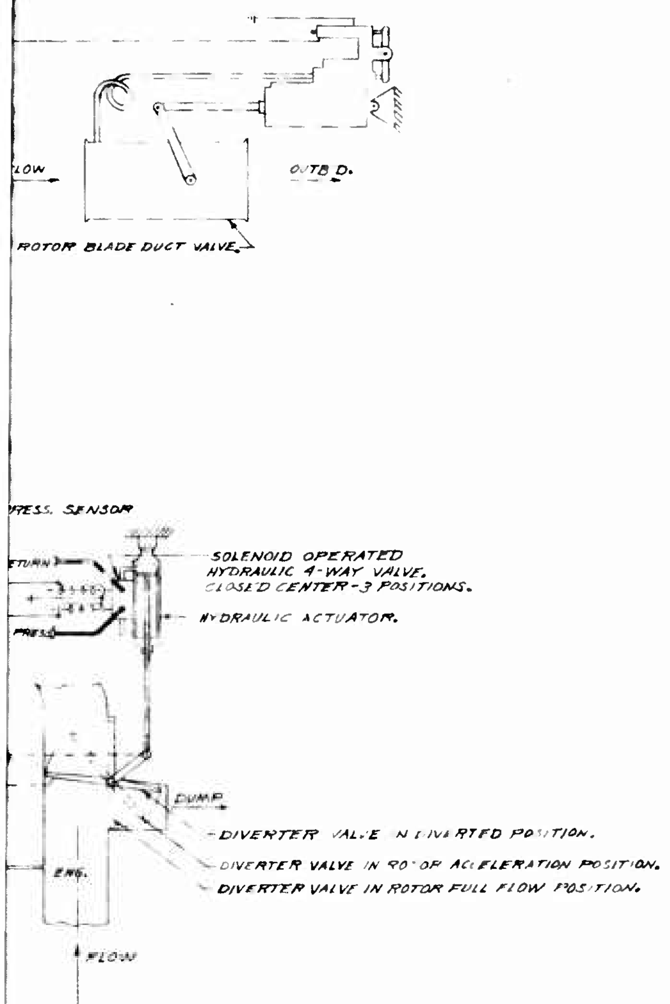



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Figure 4-14
Report 285-19 (62-19)

POSITIONS
OSITION OF DIVERTER VALVE.
THERS.
TED POSITION OF DIVERTER VALVE.
THERS.
$P_R$ =0 BETWEEN THE TWO ENGINES.
A PRE-DETERMINED VALUE.
LOSED.
IS OPEN.

| REQD | PART NO. | REQD | PART NO. | NAME | SIZE | DESCRIPTION | SPECIFICATION |
|---|---|---|---|---|---|---|---|
| ASSEMBLY OPP. | | ASSEMBLY SHOWN | | LIST OF MATERIAL | | | |

| NEXT ASSY | USED ON | NEXT ASSY | FINAL ASSY | UNLESS OTHERWISE SPECIFIED | | | |
|---|---|---|---|---|---|---|---|
| APPLICATION | | QTY REQD | | DIMENSIONAL TOLERANCES 3 PLACE DECIMAL ± .010 2 PLACE DECIMAL ± .03 ANGULAR ± 0°30' DIMENSIONS TO BE MET BEFORE PLATING. CORNER RADIUS .062 ON C' BORES AND SPOT FACES OF 1.250 DIA. OR LESS — .093 RADIUS ON GREATER THAN 1.250 DIA. | | | |

| | | |
|---|---|---|
| DRWN Sallows | 2/15/62 | SCHEMATIC – HOT CYCLE GAS CONTROL SYSTEM. |
| CHK'D | | |
| APP'D | 2/15/62 | |
| APP'D | | |
| APP'D | | |
| APP'D | | |
| APP'D | | |
| APP'D | | SCALE —— |

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

285-0008
CODE 02731 SHEET OF

## SECTION 5

## OPERATION WITH ENGINES OF UNEQUAL QUALITY

All of the engine-rotor control study was conducted on the premise that both gas generators in a two-engine installation are engines of equal quality. In this sense, equal quality is meant to be equal compressor efficiency, combustion efficiency and pressure drop, and gas generator turbine efficiency. If these factors are all equal, the two gas generators will produce the same discharge total pressure, mass flow, discharge temperature, and gas generator speed at the same fuel flow. Because these two engines are discharging into a common plenum, static pressure will be equal, and total pressures will also be equal if engine qualities are equal.

In the practical case, the engines will be of somewhat different quality, even within the guarantees of the engine for new engines. Old engines that have more operating time are likely to display even more difference in performance. This difference will be evidenced in several ways:

5.1 In equilibrium, at equal discharge pressure, the poor engine will use more fuel than the good engine. If the quality difference between the engines is sufficient, the poorer engine may hit the fuel acceleration limit while the better engine is operating within normal condition. Conceivably, this difference between the engines could be enough to bring on an unstable situation where the poorer engine will be unable to match the discharge pressure of the good engine because of the acceleration fuel limit mentioned here. The poor gas generator may then be driven to surge. The potential problem will be explored in preliminary tests on the whirl tower, as discussed below.

5.2 During power bursts, the good engine may respond faster to the added fuel than the poorer engine. This will lead perhaps to much more <u>transient</u> difference in discharge pressure than would be evidenced at equilibrium. The fuel limit may be reached by the poorer engine during a transient, and the same unstable condition of the good engine driving the poorer engine to surge may result. The transient effects of difference in quality can be minimised by applying power increments at a slower rate. However, it may be unacceptable to a pilot to apply the power so slowly that transient effects of quality differences are minimized.

The examples cited here are the extreme that could be encountered. It is felt that the actual difference between engines would amount to only

2-3% in over-all behavior, or not enough to reach fuel or gas generator speed limits that could cause one engine to drive another down to idle. In Reference 18, information is presented of the effects of having two different quality J79-8's in parallel driving two X378 lift fans through a common duct. Compressor efficiency was lowered two percent and the turbine efficiency was lowered one percent to obtain the poor engine. The case of military power was studied with constant temperature between engines and then with constant gas generator speed.

In the constant temperature case, in equilibrium, the good gas generator will run overspeed and produce 1% more mass flow than the poor engine. In the constant speed case, the good gas generator will run $17^{\circ}R$ over temperature and the poor engine will run $68^{\circ}R$ over temperature. The poor engine requires 5% more fuel flow. In either case, the poor engine does not seem to be so seriously influenced by speed or temperature limit that the poor engine is shut off by the good engine. Therefore, it appears that moderate differences of quality can exist between engines discharging into a common plenum, and equilibrium can be obtained. It is not known if an unstable condition will result for larger differences in quality. The present report includes a test program for pre-whirl test of the two T64 gas generators that will be used to drive the hot cycle rotor. This test program is given below. It is designed to demonstrate quickly and economically the transient and equilibrium effect of any quality differences between the specific T64 gas generators made available to this contract.

## SECTION 6

## PROPOSED PRE-WHIRL TESTS OF GAS GENERATORS ON WHIRL TOWER

This Contractor was unable to discover any information concerning the operation of two turbojet gas generators into a common duct that led to a common exhaust nozzle. Studies of similar installation (such as Reference 18) performed by an experienced manufacturer indicates no serious operational or control problem, at least as long as the engines are nearly equal in quality. However, it is possible that the gas generators made available to this Contractor may be sufficiently different in quality as to introduce unforeseen control problems. In addition, in the case of one engine failure, there will be a time when the good engine is temporarily exposed to twice its normal exhaust nozzle area while the bad engine is coasting to a stop and before the blade duct valve has had a chance to reduce the effective nozzle area to normal.

Because situations such as this involve primarily engine time response, it is possible to test the engines alone at the whirl tower with a temporary exhaust nozzle instead of with the regular rotor and blade tip exhaust nozzles. Thus potential control problems can be explored without unnecessarily risking the rotor. The temporary exhaust nozzle would be a short section of straight duct with a converging nozzle mounted directly above the lower half of the rotating seal which connects the stationary whirl tower ducting to the rotor ducting. The temporary nozzle would not rotate. It would have an effective exhaust area equal to the total of the rotor blade exhaust nozzle areas.

It is proposed to conduct the following pre-whirl tests:

### 6.1 DIVERTER VALVE TESTS

Configuration: Both diverter valves turned to "Overboard" position.

Objectives: a. Familiarize personnel with starting and shutting off one and two engines.

b. Verify that effective nozzle areas of "Overboard" nozzle puts engine on proper operating line so as to permit:

(1) Approximate check of full power conditions

(2) One engine out tests described below.

Record: Normal engine instruments.

Procedure: Operate each engine through its own diverter valve.

## 6.2 ENGINE QUALITY CHECK IN EQUILIBRIUM

Configuration: Diverter valves in "Rotor" position. Differential pressure sensor connected to diverter valves in accordance with Figure 4-14.

Objectives:

a. Determine operating conditions of each engine when both are exhausting through temporary nozzle.

b. Determine feasibility of matching engine discharge total pressure by making any signal to Differential Pressure Sensor go to zero.

c. Determine if matching discharge total pressure forces either engine to fuel schedule limit or gas generator speed limit at any power setting.

Record: Normal engine instruments plus output of Differential Pressure Sensor.

Procedure:

a. Start both engines with diverter valves in "Overboard" position. Place power control lever at "IDLE."

b. Move both diverter valves together to "Rotor" position.

c. Move both power control levers toward maximum power slowly.

d. Observe Differential Pressure Sensor signal and try to make signal go to zero by matching one engine pressure against the other.

e. Watch for either engine reaching any limits as prescribed by engine manufacturer.

Note: If either engine shows a malfunction, move both power control levers to "IDLE" and move both diverter valve controls to "Overboard" position.

## 6.3 ONE ENGINE OUT OPERATION

Configuration: Diverter valves in "Rotor" position.

Objectives: Determine behavior of one gas generator with twice normal effective nozzle area.

Record: Engine instruments, differential pressure sensor, diverter valve position.

Procedure:

a. Start both engines with diverter valves in "Overboard" position. Place power control levers on "IDLE."

b. Move both diverter valves together to "Rotor" position.

c. Move No. 1 diverter valve control back to "Overboard" position. Observe gas generator speed on engine with two times nozzle area. If No. 2 gas generator speed is too high, turn No. 2 diverter valve to "Overboard" position.

d. If No. 2 gas generator speed is not above limit, return No. 1 diverter valve control to "Rotor" position. This action will cause gas from No. 1 engine to help fill up the temporary nozzle and simulate the closing action of the blade duct valves. (The time constant of the return of the No. 1 diverter valve should be made equal to the time constant of the blade duct valve for this test.)

e. If No. 1 and No. 2 gas generators return satisfactorily to equilibrium, repeat steps b, c, and d with successively higher power settings above "IDLE."

f. At high power settings, transient No. 2 gas generator speed may reach or exceed limits, so decision must be made to reduce power and return No. 1 diverter to "Rotor" position or to turn No. 2 diverter to "Overboard." It is hoped, however, that the fuel control of No. 2 engine will reset fast enough to keep No. 2 speed within limits. Otherwise, corrective action must be taken.

g. After exploring one engine out (up to full power if possible) by reversing diverter valve positions, a check must be made of operation of the Differential Pressure Sensor. This is done simply by starting with both diverter valves in the "Rotor" position and power control levers at the same moderate power setting. By advancing one power control lever ahead of the other, a signal will be produced of 2 psi or greater. This signal should cause the opposite diverter valve to more to the "Overboard" position. The two engines should be matched again by retarding the power control lever that had been advanced back to match the one that was not moved. This action should cause the diverter valve that had moved to return to "Rotor" position.

## 6.4 POWER RECOVERY QUALITY CHECK

Configuration: Diverter Valves in Rotor Position

Objective: To set threshold $\Delta P$ signal of differential pressure sensor high enough so differences in quality of engines do not act to shut off a good engine during an accelerated power recovery.

Record: All instrumentation plus diverter valve position.

Procedure: a. Set differential pressure threshold signal at 4 psi.

b. With diverter valves set to "Rotor" position and power control levers at "IDLE," advance power control levers rapidly to full setting.

c. If any transient difference in pressure that exists during acceleration does not cause differential pressure sensor to divert one engine, reset threshold signal to 3 psi and repeat a. and b..

Note: Minimum signal with one engine idling at sea level is 3 psi.

## SECTION 7

## CONCLUSIONS

A study has been made of the engine-rotor control system for a hot cycle rotor powered by two General Electric T64 gas generators with the following results:

1. The basic governor of the YT64 turboshaft engine can successfully govern the hot cycle helicopter rotor without modifications.

2. The static and dynamic percent droop of rotor RPM following a load change will be of the same order as, or less than, the values experienced with other current free turbine turboshaft engines in single or dual installations.

3. The maximum frequency response of the engine-rotor control system is well below even the lowest possible aerodynamic forcing frequency of one per rev; therefore, there will be no large oscillatory variations in rotor RPM due to external excitation.

4. If two engines are initially at 50% power and one engine fails, the remaining good engine will accelerate to full power within two seconds. The transient RPM droop will be 5%. The steady RPM droop will be 4%, and it can be reduced to zero by resetting the governed RPM.

5. If the pilot makes a power recovery from autorotation, full rotor power will be obtained in 3 seconds and initial rotor RPM will be obtained in about six seconds. These times are comparable to the behavior of other free turbine installations.

6. A preliminary design study was made of the requirements for diverter valves and blade duct valves to permit operation with one engine out. The study involved the following components:

   a. Three gas generator-diverter valve configurations.

   b. Six diverter valve designs, including the available General Electric J85 diverter valve.

   c. Two blade root duct valves and one blade tip valve.

7. Any of these components could be used, depending on the helicopter configurations, with little effect on the dynamics of the over-all

system; a more detailed design study will be required to select the optimum combination of components for a particular application.

8. A detailed discussion of the operation of the components is given; some operations are necessarily automatic and some can be performed safely by the pilot with consequent simplification of the control system.

9. A brief test program is presented for pre-whirl tests of the gas generators on the whirl tower to demonstrate operation of the two engines through a common duct before exposing the rotor to possible damage.

## SECTION 8

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## APPENDIX I

## DETAILED OPERATION OF CONTROL SYSTEM

(REFER TO SCHEMATIC DIAGRAM - FIGURE 4-14)

The diverter valves will be actuated by dual hydraulic actuating cylinders of conventional design, pressure being available from the 3000 psi dual hydraulic system provided for the primary flight controls. Dual solenoid operated three position hydraulic four-way valves will be provided. The dual valves will be mechanically inter-locked so that energizing of the solenoids on either valve will operate both valves. Provision of completely duplicated hydraulic and electric power supplies, plumbing, wiring, actuators and valves will assure an extremely high level of reliability and virtually rule out the possibility of failure of the valve actuating cylinders.

The four-way valves will be "closed-center" type, which means that in their center or "neutral position" all ports are blocked. When all solenoids are de-energized the valves will be spring centered and the cylinders will be hydraulically locked in position.

The blade duct valves will be actuated to their closed position by single acting hot-gas cylinders, utilizing pressure tapped directly from the blade ducts. Solenoid operated valves will be used to control the position of the blade duct valves. The solenoids will operate poppet type valves through rocker-arms. Their masses and lever arms will be such that the centrifugal forces on the solenoid plungers and the poppet valves will be in balance in order to minimize the operating loads for the solenoids.

The poppet valves will be the "normally closed" three-way type. When the solenoid is de-energized the poppet will be held on the inlet seat and the hot-gas cylinder will be open to the exhaust port forcing the duct-valve to its open position under centrifugal force loading. When the solenoid is energized the poppet will be forced to the seat on the exhaust side of the valve, admitting pressure to the actuating piston. This will pull the tip valves to their closed positions against the pressure and centrifugal forces which normally hold them in the open position. Dual solenoids and duplicate wiring and switches will be used throughout for reliability. Separate hot gas actuators and solenoid valves will be provided for each of the three blade duct valves.

The hot gas cylinders and valves will be of all metal construction, including seals. No moving seals will be required for the valves. Conventional piston-rings can be used on the actuating pistons. Operating pressures and forces are low and stresses will therefore be inherently low since these units will have to be designed with practical minimum wall thicknesses. Although gas temperatures will be high the ambient temperature on these units will be nominal. When pressure is holding the duct valve in closed position the only gas flow through the actuator will be leakage by the piston rings, which will be very low for this low pressure application. Good heat dissipation can

therefore be expected.

Dry-film lubricants have been successfully applied to hot gas valves and actuators of this type previously developed and produced by the Hughes Tool Company and their use is intended for this application.

The solenoids will be continuous duty D. C. type and will be exposed to fairly moderate temperatures. These are also well within the area of past experience at this company.

Referring to the schematic diagram, operation of the circuit will be as follows:

The pilot's diverter valve controls will consist of a three position switch for each of the two engine diverter valves. For engine starting and shut-down these switches will be placed in position "A". This will energize the circuits to the proper solenoids on the hydraulic four-way valves for pressurizing the hydraulic pistons in the direction to hold the diverter valves in the diverted flow position, as shown on the schematic diagram of Figure 4-14. Internal stops will be provided in the hydraulic actuators to prevent the full hydraulic load from being applied to the diverter valve seals.

After the engines are started and it is desired to accelerate the rotor the diverter valve controls will be moved to the "accelerate" position, "B". This will de-energize one set of solenoids on the hydraulic four-way valves and energize the other set to cause hydraulic pressure to be ported to the opposite sides of the pistons, moving the diverter valves in the direction to open flow to the rotor. At an intermediate position in which part of the gas flow is diverted and part admitted to the rotor blades, cams in the diverter valve linkages will operate micro-switches (1), opening the circuits. Thus, both sets of solenoids will be de-energized and the hydraulic four-way valves will return to their neutral positions, hydraulically locking the diverter valves in this position. At all other positions of the diverter valve micro-switches (1) will be closed. Micro-switches (2) will have opened as soon as the diverter valves were moved from the diverted position.

It should be noted that with the diverter valve controls in either positions "A" or "B" there is no electrical circuit to the rotor and therefore the duct valve solenoids are de-energized. Therefore there will be no gas pressure on the tip valve actuator in either of these positions.

After both engines are started and synchronized the differential total pressure between them will be close to zero. Thus the sensing element in the differential pressure switch will be centered, maintaining micro-switch contacts (3) on both sides closed and contacts (4) open.

After acceleration of the rotors the diverter valve controls will both be placed in position "C". This will energize the circuits to the proper solenoids on the hydraulic four-way valves to move and hold the diverter valves in the "Rotor" position. Normal flight operation will be in this condition. Microswitches (2) are open in this condition and therefore in normal flight the blade duct valve solenoids are de-energized and the duct-valves are held in their open position.

The following sequence of operations will take place in the event of single engine failure:

When the differential total pressure between the two engines exceeds 3 psi a power failure will be indicated. The differential pressure sensor will then operate to open micro-switch contact (3) on the side connected to the failed engine and close contact (4). This will reverse the circuits to the solenoids on the hydraulic four-way valves to cause hydraulic pressure to move the proper diverter valve to the diverted position. When this position is reached, micro-switch (2) will close, energizing the circuit to the duct valve solenoids through the slip ring contacts. This will cause the blade duct valves to move to their closed positions.

The diverter valve controls will be left in position "C" under the condition described. If the inoperative engine is restarted the differential total pressure between the two engines approaches zero. The differential pressure sensor will then cause micro-switch contact (4) to open and (3) to close, again reversing the circuits to the solenoids on the hydraulic four-way valves. Thus, the diverter valve will automatically return to the "Rotor" position. As soon as the diverter valve leaves the diverted flow position, micro-switch (2) will open, de-energizing the circuits to the blade duct valve solenoids, so that the blade duct valves will return to their open positions. The time constant on the duct-valves will be substantially lower than the diverter valve time constant to insure that the duct valves are in the open position well before the diverter valve admits full engine flow to the blade ducts.

NOTE - The above description is predicated on the use of blade duct valves at the tips of the rotors. If the valves were at the roots of the rotor blade, the operation would be similar; however in this case "normally open" solenoid valves would be used and duct pressure on the actuating pistons would hold the valves open. Centrifugal force would be used to close the root duct valves.